AEROSPACE REPORT NO.
ATR-2003(7949)-2

National Oceanic and Atmospheric Administration (NOAA) Geostationary Operational Environmental Satellite-R (GOES-R) Concept Design Center Space Segment Team (SST064) Study 3

30 September 2003

Prepared by

THE CONCEPT DESIGN CENTER
Space Segment Team
Engineering and Technology Group

Prepared for

U.S. DEPT. OF COMMERCE, NOAA
Silver Spring, MD 20910

Contract No. 50-SPNA-0-00012

Civil & Commercial Division

THE AEROSPACE CORPORATION
EL SEGUNDO, CALIFORNIA

AEROSPACE REPORT NO.
ATR-2003(7949)-2

NATIONAL OCEANIC AND ATMOSPHERIC ADMINISTRATION (NOAA) GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE-R (GOES-R) CONCEPT DESIGN CENTER SPACE SEGMENT TEAM (SST064) STUDY 3

Prepared by

THE CONCEPT DESIGN CENTER
Space Segment Team
Engineering and Technology Group

30 September 2003

Civil & Commercial Division
THE AEROSPACE CORPORATION
El Segundo, CA 90245-4691

Prepared for

U.S. DEPT. OF COMMERCE, NOAA
Silver Spring, MD 20910

Contract No. 50-SPNA-0-00012

AEROSPACE REPORT NO.
ATR-2003(7949)-2

NATIONAL OCEANIC AND ATMOSPHERIC ADMINISTRATION (NOAA) GEOSTATIONARY OPERATIONAL ENVIRONMENTAL SATELLITE-R (GOES-R) CONCEPT DESIGN CENTER SPACE SEGMENT TEAM (SST064) STUDY 3

Prepared

J. A. Aguilar
Vehicle Concepts Department
Systems Engineering Division
Engineering and Technology Group

Approved

F. Knight, Director
Vehicle Concepts Department
Systems Engineering Division
Engineering and Technology Group

J. Soukup, Senior Project Leader
NOAA/NESDIS Office of Systems Development
Civil and Commercial Division

Abstract

The Aerospace Corporation's Concept Design Center Space Segment Team performed a Geostationary Operational Environment Satellite-R (GOES-R) study for the National Oceanic and Atmospheric Administration (NOAA). During this study, which took place January 14–16, 2003, 13 spacecraft configurations were developed for the next-generation GOES. These spacecraft configurations were based on previous designs conducted in an earlier study [National Oceanic and Atmospheric Administration Geostationary Operational Environmental Satellite-R Concept Design Center Space Segment Team Study, ATR-2003(7949)-1]. Three different spacecraft architectures were explored with these spacecraft designs: (1) a consolidated spacecraft architecture, (2) a distributed spacecraft architecture, and (3) a MEO spacecraft architecture. This document contains the results of the study, including issues identified and recommendations.

Contents

Figures

Tables

1. Introduction

Joseph Aguilar

1.1 Study Background

A third Concept Design Center (CDC) Space Segment Team (SST) study performed for the National Oceanic and Atmospheric Administration (NOAA) took place January 14–16, 2003 in The Aerospace Corporation's real-time design facility in El Segundo, California. The purpose of the study was to continue to examine the spacecraft concepts developed in the previous study [National Oceanic and Atmospheric Administration Geostationary Operational Environmental Satellite-R Concept Design Center Space Segment Team Study, ATR-2003(7949)-1].

The Aerospace Corporation's CDC is a real-time design facility bringing together all of the subsystem experts needed to design a spacecraft. Each subsystem "seat" runs various software models to capture requirements, performance, and impacts to the spacecraft. This information is then linked in real time to the other subsystems. Through an iterative process, the models will converge to a point design satisfying all of the given requirements for a particular configuration. Since the customer is present during these sessions, active discussion with the subsystem seats provides feedback to the customer, and allows the customer to clarify any requirements questions that may arise. It should be noted that, at the request of the customer, the cost, ground, and software subsystems were not used in this study.

A few days prior to the start of the design sessions, the study leads discussed spacecraft requirements with the customer, goals for the study, and pertinent configurations to be examined. Subsystem experts were brought in where necessary to lend expertise to the pre-session decisions and to perform any pre-work that would allow the sessions to run more smoothly.

1.2 Mission Overview

The GOES program is a key element in National Weather Service (NWS) operations, providing a continuous stream of environmental information (weather imagery and sounding data) used to support weather forecasting, severe storm tracking, and meteorological research. From their geosynchronous positions over the eastern U.S./Atlantic and western U.S./Pacific, the two spacecraft can "stare" at most of the western hemisphere to provide cloud images, Earth surface temperatures, water vapor fields, and vertical thermal and vapor structures. This data is used to follow the evolution of atmospheric phenomena, ensuring real-time coverage of short-lived dynamic events, especially severe local storms and tropical cyclones—two meteorological events that directly affect public safely, protection of property, and ultimately, economic health and development. – *GOES I-M DataBook*

1.3 Team Members

The CDC SST members that supported this GOES-R design study are listed in Table 1.1. The team was formed from members of The Aerospace Corporation's technical staff who were selected to provide broad technical expertise and experience.

Table 1.1. CDC Team Members

Subsystem	Team Member
ADACS	Andrei Doran, Elias Polendo
Astrodynamics	Tom Lang
C&DH	Ron Selden
Configuration	Scott Szogas
Payload Communications	John O'Donnell
Power	Ed Berry
Propulsion	Keith Coste
Structures	Kenneth Mercer
Study Leads	Joseph Aguilar, Ron Bywater
Systems	Alice Moke
Thermal	Bill Fischer
TT&C	John O'Donnell

1.4 Customers

Mike Crison, Director of Requirements and Systems Programs in the Office of Systems Development at NOAA NESDIS, represented the interests of NOAA and was the primary customer for this study. There were a number of additional participants from NOAA, NASA, and other interested agencies. Jim Soukup, Senior Project Leader in the Civil and Commercial Division supporting NOAA NESDIS, was the Aerospace customer interface.

1.5 Disclaimer

This report constitutes the results of the CDC study. It is intended to assess feasibility and to estimate the required technologies, equipment, mass, and deployment strategy required to implement the customer's mission goals. The designs documented herein are intended to be conceptual solutions, developed with a minimum expenditure of work force and time. As a result, these representative solutions have not been optimized and may be incomplete or vary significantly from eventual systems. It is strongly recommended that a more detailed study be completed before final implementation decisions are made. Some more detailed analysis in the following areas is recommended:

- In-depth field-of-view (FOV) analysis should be performed to verify that all of the sensors and antenna have adequate clear field-of-regard.
- Further design and analysis of the stacked spacecraft configurations should be performed.
- Detailed trade study should be performed to determine hardware design and approach for payload and command and data handling interface.
- Detailed study should be made of jitter sources, requirements, and solutions.

- Further design and analysis of the medium-Earth orbit (MEO) spacecraft concept should be performed.
- Command and data handling mass memory capacity required should be further explored.
- More detailed structural disturbance analysis should be performed.

This page intentionally blank.

2. Payload

Joseph Aguilar

2.1 Payload Summary

There were several payloads used over the course of the 13 different spacecraft configurations. Table 2.1 summarizes the payloads used.

All of these payloads had a duty cycle of 100% in both daylight and eclipse, with one exception. The SXI payload did not operate at all during eclipse, yet it did use 50 W during eclipse for standby power. The SXI payload data also includes the solar coronagraph payload data.

Table 2.1. Payload Summary

Payload	Mass (kg)	Power (W)	Data Rate
ABI	275	450	60 Mb/s
Aux 1	17.9	29.7	1 Mb/s
Aux 2	17.9	29.7	1 Mb/s
DCS	17.9	29.7	Low B/s
EHS	180	190	1.2 Mb/s
FDS	157	300	65 Mb/s
GEOSTAR	100	250	2 Mb/s
GMS	300	300	500 kb/s
HES-1	200	450	65 Mb/s
HES-2	80	100	2.6 Mb/s
Lightning Mapper	37.5	144	200 kb/s
RHS	80	100	1.4 Mb/s
SAR	8.6	22.4	Low B/s
SEM	54	94	560 B/s
SXI	50	200	2.8 Mb/s

2.2 Payload Acronyms

ABI	Advanced Baseline Imager
DCS	Data Collection System
EHS	Emissive Hyperspectral Sounder
FDS	Full Disk Sounder
GMS	Geostationary Microwave Sounder
HES	Hyperspectral Environmental Suite
MFS	Multi-function Sensor
RHS	Reflective Hyperspectral Sensor
SAR	Search and Rescue
SEM	Space Environment Monitor
SXI	Solar X-ray Imager

This page intentionally blank.

3. Systems

Alice Moke

3.1 Requirements

For this study, 3 different architectures were explored: (1) the ABC architecture, (2) the consolidated spacecraft architecture, and (3) the MEO spacecraft architecture. The A Sat, B Sat, and C Sat spacecraft were used for the ABC architecture. The AB Sat and MEO Sat spacecraft were used for the consolidated and MEO spacecraft architecture, respectively. All but two of the 13 spacecraft configurations designed during this study, were designed to operate in a geosynchronous-Earth orbit (GEO). Table 3.1 shows the guidelines followed in designing these spacecraft.

Two of the 11 GEO spacecraft configurations were designed after the study. These configurations, Configuration 12 and Configuration 13, followed the same guidelines shown in Table 3.1 with one exception. The spacecraft lifetime for these two configurations is 15 years, all on-orbit.

Two of the spacecraft configurations were designed to operate in a MEO at 10,385 km at 0° inclination. The MEO spacecraft design requirements are shown in Table 3.2.

Table 3.1. GEO Spacecraft Guidelines

Mission	
Spacecraft Lifetime	10 years (7 operational, 2 on-orbit spare, 1 ground spare)
Ground Lifetime	16 years
Launch Date	2012
Technology Freeze Date	2008
Mission Orbit	GEO, 75° E and 137° W
Inclination Tolerance	±0.5°
Desired Launch Vehicle	EELV Medium
Constellation Size	1 Spacecraft Cluster at Each Orbit Slot
Spacecraft	
Redundancy	Full
Heritage	Commercial
Stabilization	3-axis
Reposition Requirements	8 in lifetime (6 @ 1°/day, 2 @ 3°/day)
Slew Requirments	Bi-annual Yaw Flip
Knowledge	7 μrad Goal, 14 Threshold
Pointing	150 μrad
Environment	Natural

Table 3.2. MEO Spacecraft Guidelines

Mission	
Spacecraft Lifetime	12 years
Ground Lifetime	16 years
Launch Date	2012
Technology Freeze Date	2008
Mission Orbit	10,385 km
Inclination	0°
Desired Launch Vehicle	EELV Medium
Constellation Size	4
Spacecraft	
Redundancy	Full
Heritage	Commercial
Stabilization	3-axis
Reposition Requirements	2 in lifetime (30° @ 3°/day)
Knowledge	175 µrad
Pointing	450 µrad
Environment	Natural

In order to relate the technological maturity and technological risk to the uncertainty of the cost estimation elements, a numeric scale has been applied to many of the subsystems and components in the design. The numeric scale, referred to as the technology readiness level (TRL), was developed by NASA and has been adapted for use in this application. Figure 3.1 shows the relation between typical programmatic phases and the level of development that a technology has received. Note that in many cases the figure distinguishes between ground (G) and space (S) experience.

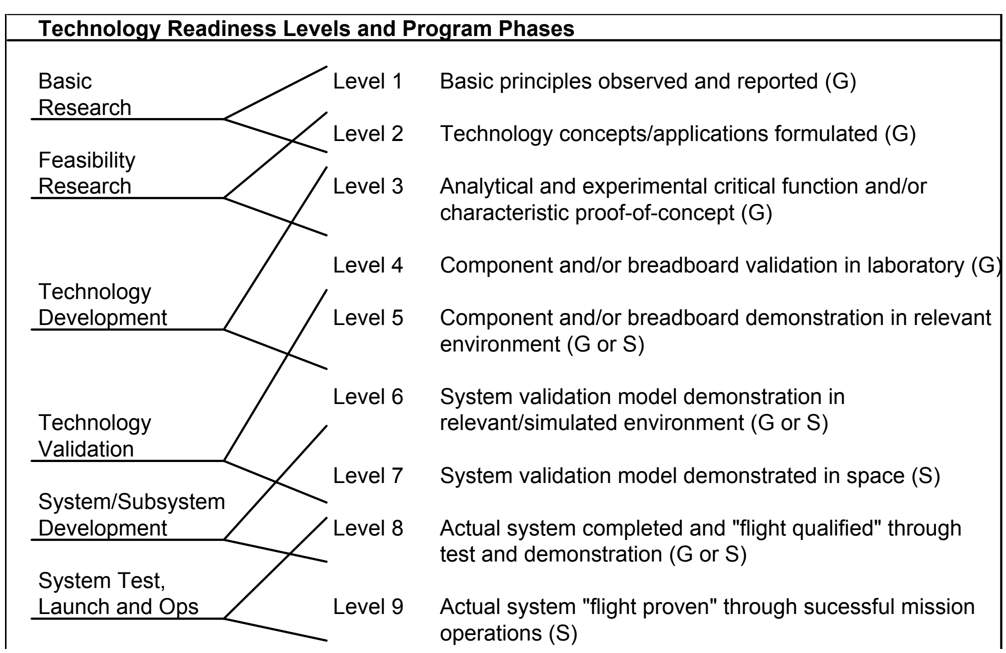

Figure 3.1. Technology readiness levels.

3.2 Spacecraft Configurations

Thirteen different spacecraft configurations were designed over the course of the design sessions. Multiple payload configurations were investigated during the course of the study. A brief description of each of the configurations is summarized in the following paragraphs. Mass and power results for each configuration are shown in Tables 3.3 to 3.15.

3.2.1 Configuration 1: Baseline A Sat

The baseline A Sat spacecraft carries the ABI, SXI, SAR, DCS, low-rate services, and global rebroadcast (GRB) payloads. One baseline A Sat spacecraft and one baseline B Sat spacecraft are launched together on an Atlas V 551 launch vehicle.

3.2.2 Configuration 2: Baseline A Sat Minus Communications

This spacecraft is identical to Configuration 1 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. One baseline A Sat minus communications spacecraft and onebaseline B Sat minus communications spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.3 Configuration 3: Baseline B Sat

The baseline B Sat spacecraft carries the HES-1, HES-2, SEM, SAR, DCS, low-rate services, and GRB payloads. One baseline A Sat spacecraft and one baseline B Sat spacecraft are launched together on an Atlas V 551 launch vehicle.

3.2.4 Configuration 4: Baseline B Sat Minus Communications

This spacecraft is identical to Configuration 3 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. One baseline A Sat minus communications spacecraft and one baseline B Sat minus communications spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.5 Configuration 5: B Sat MIT

Similar to Configuration 3, this configuration replaces the HES-1 and HES-2 payloads with the FDS, RHS, and EHS payloads. One baseline A Sat spacecraft and one B Sat MIT spacecraft are launched together on an Atlas V 551 launch vehicle.

3.2.6 Configuration 6: Baseline C Sat

The baseline C Sat spacecraft carries the Lightning Mapper, GEOSTAR, SAR, DCS, low-rate services, and GRB payloads. One baseline C Sat spacecraft and one baseline A Sat minus communications spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.7 Configuration 7: Baseline C Sat Minus Communications

This spacecraft is identical to Configuration 6 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. One baseline C Sat minus communications spacecraft and one baseline A Sat spacecraft are launched together on an Atlas V 531 launch vehicle.

3.2.8 Configuration 8: Baseline AB Sat

The baseline AB Sat spacecraft carries the ABI, SXI, HES-1, HES-2, SEM, SAR, DCS, low-rate services, and GRB payloads. One baseline AB Sat spacecraft is launched on an Atlas V 531 launch vehicle.

3.2.9 Configuration 9: Baseline AB Sat Minus Communications

This spacecraft is identical to Configuration 8 except that the low-rate services, GRB, SAR, and DCS payloads have been removed. Two baseline AB Sat spacecraft are launched together on a Delta IV Heavy launch vehicle.

3.2.10 Configuration 10: Baseline MEO Sat

The baseline MEO Sat spacecraft carries the GEOSTAR, Aux 1, Aux 2, SAR, DCS, low-rate services, and GRB payloads. Four baseline MEO Sat spacecraft are launched on a Delta IV Heavy launch vehicle.

3.2.11 Configuration 11: Baseline MEO Sat GEO to MEO Insertion

The payloads for this spacecraft design are the same as those for Configuration 10. The design of the spacecraft, however, is different. This spacecraft is inserted into a GEO orbit and uses the on-board propulsion system to get into the MEO.

3.2.12 Configuration 12: A Sat D1

This spacecraft is identical to the baseline A Sat spacecraft with two exceptions. First, the SXI payload is removed. Second, the spacecraft design life is 15 years, all on-orbit. One A Sat D1 spacecraft and one B Sat D2 spacecraft are launched together on an Atlas V 551 launch vehicle.

3.2.13 Configuration 13: B Sat D2

This spacecraft is identical to the baseline B Sat spacecraft with two exceptions. First, the SEM payload is removed. Second, the spacecraft design life is 15 years, all on-orbit. One A Sat D1 spacecraft and one B Sat D2 spacecraft are launched together on an Atlas V 551 launch vehicle.

Table 3.3. Configuration 1: Baseline A Sat

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**537.9**	**1186.0**	**43%**	**1269.6**	**1119.6**	
Custom Payload	450.0	992.3		900.0	750.0	5
Payload Communications	70.3	155.0		295.7	295.7	5
Payload Contingency	17.6	38.7		73.9	73.9	
Spacecraft	**894.5**	**1972.3**		**559.0**	**559.0**	
Propulsion	136.6	301.1	11%	0.2	0.2	7
ADACS	65.0	143.3	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	34.9	76.9	3%	237.1	237.1	9
Power	177.5	391.5	15%	0.0	0.0	6
Structure	246.1	542.5	20%	0.0	0.0	6
Spacecraft Contingency	206.4	455.2		129.0	129.0	
Satellite Summary						
EOL Power				**1828.7**	**1678.7**	
BOL Power				**2706.9**		
Dry Mass	**1432.3**	**3158.3**				
Orbit Insertion Propellant	1683.5	3712.2				
On-Orbit Propellant	480.3	1059.1				
Pressurant	5.4	12.0				
Wet Mass	**3601.6**	**7941.6**				

Table 3.4. Configuration 2: Baseline A Sat Minus Communications

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**446.9**	**985.4**	**42%**	**859.9**	**709.9**	
Custom Payload	400.0	882.0		750.0	600.0	5
Payload Communications	37.5	82.8		87.9	87.9	5
Payload Contingency	9.4	20.7		22.0	22.0	
Spacecraft	**784.5**	**1729.9**		**503.8**	**503.8**	
Propulsion	136.6	301.1	13%	0.2	0.2	7
ADACS	59.0	130.1	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	30.1	66.3	3%	194.6	194.6	9
Power	136.0	299.9	13%	0.0	0.0	6
Structure	213.8	471.5	21%	0.0	0.0	6
Spacecraft Contingency	181.0	399.2		116.3	116.3	
Satellite Summary						
EOL Power				**1363.7**	**1213.7**	
BOL Power				**2041.0**		
Dry Mass	**1231.5**	**2715.4**				
Orbit Insertion Propellant	1456.8	3212.2				
On-Orbit Propellant	421.1	928.5				
Pressurant	4.7	10.4				
Wet Mass	**3114.0**	**6866.4**				

Table 3.5. Configuration 3: Baseline B Sat

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**537.9**	**1186.0**	**43%**	**1269.6**	**1269.6**	
Custom Payload	450.0	992.3		900.0	900.0	5
Payload Communications	70.3	155.0		295.7	295.7	5
Payload Contingency	17.6	38.7		73.9	73.9	
Spacecraft	**901.3**	**1987.5**		**560.4**	**560.4**	
Propulsion	136.6	301.1	11%	0.2	0.2	7
ADACS	65.0	143.3	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	35.0	77.3	3%	238.1	238.1	9
Power	181.7	400.5	15%	0.0	0.0	6
Structure	247.1	544.8	20%	0.0	0.0	6
Spacecraft Contingency	208.0	458.6		129.3	129.3	
Satellite Summary						
EOL Power				**1830.0**	**1830.0**	
BOL Power				**2723.4**		
Dry Mass	**1439.2**	**3173.5**				
Orbit Insertion Propellant	1691.0	3728.7				
On-Orbit Propellant	482.4	1063.8				
Pressurant	5.4	12.0				
Wet Mass	**3618.1**	**7977.9**				

Table 3.6. Configuration 4: Baseline B Sat Minus Communications

	Mass			Power [W]		NASA
	[kg]	[lbs]	%dry	Daylight	Eclipse	TRL
	Time (min) -->			1366.7	69.4	
Payload	**446.9**	**985.4**	**42%**	**859.9**	**859.9**	
Custom Payload	400.0	882.0		750.0	750.0	5
Payload Communications	37.5	82.8		87.9	87.9	5
Payload Contingency	9.4	20.7		22.0	22.0	
Spacecraft	**791.7**	**1745.6**		**506.8**	**506.8**	
Propulsion	136.6	301.1	13%	0.2	0.2	7
ADACS	59.0	130.1	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	30.2	66.6	3%	196.9	196.9	9
Power	140.3	309.3	13%	0.0	0.0	6
Structure	214.9	473.8	21%	0.0	0.0	6
Spacecraft Contingency	182.7	402.8		117.0	117.0	
Satellite Summary						
EOL Power				**1366.7**	**1366.7**	
BOL Power				**2059.8**		
Dry Mass	**1238.6**	**2731.0**				
Orbit Insertion Propellant	1464.5	3229.3				
On-Orbit Propellant	423.3	933.4				
Pressurant	4.7	10.4				
Wet Mass	**3131.1**	**6904.2**				

Table 3.7. Configuration 5: B Sat MIT

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**635.4**	**1401.0**	**46%**	**1205.7**	**1205.7**	
Custom Payload	547.5	1207.2		836.1	836.1	5
Payload Communications	70.3	155.0		295.7	295.7	5
Payload Contingency	17.6	38.7		73.9	73.9	
Spacecraft	**933.5**	**2058.3**		**581.3**	**581.3**	
Propulsion	143.0	315.4	11%	0.2	0.2	7
ADACS	61.0	134.5	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	38.6	85.1	3%	254.2	254.2	9
Power	176.8	389.9	13%	0.0	0.0	6
Structure	270.6	596.7	20%	0.0	0.0	6
Spacecraft Contingency	215.4	475.0		134.1	134.1	
Satellite Summary						
EOL Power				**1787.0**	**1787.0**	
BOL Power				**2645.3**		
Dry Mass	**1568.8**	**3459.3**				
Orbit Insertion Propellant	1865.3	4113.0				
On-Orbit Propellant	537.8	1185.8				
Pressurant	6.0	13.3				
Wet Mass	**3977.9**	**8771.2**				

Table 3.8. Configuration 6: Baseline C Sat

	Mass [kg]	Mass [lbs]	%dry	Power [W] Daylight	Power [W] Eclipse	NASA TRL
	Time (min) -->			1366.7	69.4	
Payload	**387.8**	**855.0**	**39%**	**1043.5**	**1043.5**	
Custom Payload	314.0	692.4		746.1	746.1	4
Payload Communications	59.0	130.1		237.9	237.9	5
Payload Contingency	14.8	32.5		59.5	59.5	
Spacecraft	**746.1**	**1645.2**		**492.2**	**492.2**	
Propulsion	115.6	254.9	12%	0.1	0.1	6
ADACS	53.0	116.9	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	27.3	60.3	3%	185.8	185.8	9
Power	154.2	340.1	16%	0.0	0.0	6
Structure	195.7	431.6	21%	0.0	0.0	6
Spacecraft Contingency	172.2	379.7		113.6	113.6	
Satellite Summary						
EOL Power				**1535.7**	**1535.7**	
BOL Power				**2282.4**		
Dry Mass	**1133.9**	**2500.2**				
Orbit Insertion Propellant	1244.6	2744.4				
On-Orbit Propellant	315.3	695.2				
Pressurant	4.0	8.9				
Wet Mass	**2697.9**	**5948.8**				

Table 3.9. Configuration 7: Baseline C Sat Minus Communications

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**311.6**	**687.0**	**38%**	**765.8**	**765.8**	
Custom Payload	287.5	633.9		694.0	694.0	4
Payload Communications	19.2	42.4		57.4	57.4	5
Payload Contingency	4.8	10.6		14.4	14.4	
Spacecraft	**646.1**	**1424.7**		**455.7**	**455.7**	
Propulsion	99.5	219.5	12%	0.1	0.1	6
ADACS	51.0	112.4	6%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	23.2	51.2	3%	157.7	157.7	9
Power	127.1	280.2	16%	0.0	0.0	6
Structure	168.2	370.9	21%	0.0	0.0	6
Spacecraft Contingency	149.1	328.8		105.2	105.2	
Satellite Summary						
EOL Power				**1221.5**	**1221.5**	
BOL Power				**1850.7**		
Dry Mass	**957.7**	**2111.7**				
Orbit Insertion Propellant	1057.4	2331.5				
On-Orbit Propellant	268.0	591.0				
Pressurant	3.4	7.6				
Wet Mass	**2286.5**	**5041.8**				

Table 3.10. Configuration 8: Baseline AB Sat

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**998.7**	**2202.1**	**44%**	**2241.2**	**2091.2**	
Custom Payload	900.0	1984.5		1800.0	1650.0	5
Payload Communications	78.9	174.1		353.0	353.0	5
Payload Contingency	19.7	43.5		88.2	88.2	
Spacecraft	**1640.4**	**3617.0**		**837.0**	**837.0**	
Propulsion	224.9	495.9	10%	0.1	0.1	6
ADACS	69.0	152.1	3%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	0%	26.7	26.7	6
Thermal	87.6	193.2	4%	439.7	439.7	9
Power	295.9	652.4	13%	0.0	0.0	6
Structure	556.4	1226.8	25%	0.0	0.0	6
Spacecraft Contingency	378.5	834.7		193.1	193.1	
Satellite Summary						
EOL Power				**3078.2**	**2928.2**	
BOL Power				**4661.2**		
Dry Mass	**2639.0**	**5819.1**				
Orbit Insertion Propellant	2937.6	6477.3				
On-Orbit Propellant	671.3	1480.2				
Pressurant	9.5	21.0				
Wet Mass	**6257.4**	**13797.5**				

Table 3.11. Configuration 9: Baseline AB Sat Minus Communications

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**857.7**	**1891.3**	**48%**	**1681.5**	**1481.5**	
Custom Payload	800.0	1764.0		1500.0	1300.0	5
Payload Communications	46.2	101.8		145.2	145.2	5
Payload Contingency	11.5	25.5		36.3	36.3	
Spacecraft	**1201.2**	**2648.6**		**716.9**	**716.9**	
Propulsion	184.8	407.5	10%	0.1	0.1	6
ADACS	65.0	143.3	4%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	26.7	26.7	6
Thermal	57.5	126.7	3%	347.3	347.3	9
Power	234.5	517.1	13%	0.0	0.0	6
Structure	354.2	781.0	20%	0.0	0.0	6
Spacecraft Contingency	277.2	611.2		165.4	165.4	
Satellite Summary						
EOL Power				**2398.4**	**2198.4**	
BOL Power				**3655.8**		
Dry Mass	**2058.9**	**4539.9**				
Orbit Insertion Propellant	2316.8	5108.6				
On-Orbit Propellant	584.8	1289.4				
Pressurant	7.5	16.5				
Wet Mass	**4968.0**	**10954.4**				

Table 3.12. Configuration 10: Baseline MEO Sat

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			316.0	44.0	
Payload	**396.0**	**873.1**	**43%**	**1130.1**	**1130.1**	
Custom Payload	312.3	688.6		661.5	661.5	4
Payload Communications	66.9	147.6		374.9	374.9	5
Payload Contingency	16.7	36.9		93.7	93.7	
Spacecraft	**659.1**	**1453.3**		**421.9**	**421.9**	
Propulsion	21.9	48.3	2%	0.9	0.9	7
ADACS	38.0	83.8	4%	95.3	95.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	25.6	56.4	3%	173.9	173.9	9
Power	209.4	461.8	24%	0.0	0.0	6
Structure	184.0	405.8	21%	0.0	0.0	6
Spacecraft Contingency	152.1	335.4		97.4	97.4	
Satellite Summary						
EOL Power				**1552.0**	**1552.0**	
BOL Power				**3378.8**		
Dry Mass	**1055.1**	**2326.4**				
Orbit Insertion Propellant	0.0	0.0				
On-Orbit Propellant	45.0	99.1				
Pressurant	0.1	0.2				
Wet Mass	**1100.1**	**2425.8**				

Table 3.13. Configuration 11: Baseline MEO Sat GEO to MEO Insertion

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			316.0	44.0	
Payload	**396.1**	**873.5**	**38%**	**1091.1**	**1091.1**	
Custom Payload	312.3	688.6		661.5	661.5	4
Payload Communications	67.1	147.9		343.7	343.7	5
Payload Contingency	16.8	37.0		85.9	85.9	
Spacecraft	**815.4**	**1798.1**		**451.7**	**451.7**	
Propulsion	116.6	257.1	12%	0.2	0.2	7
ADACS	38.0	83.8	4%	95.3	95.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	11.2	24.7	1%	15.3	15.3	6
Thermal	29.1	64.1	3%	197.5	197.5	9
Power	208.4	459.4	21%	0.0	0.0	6
Structure	207.2	456.9	21%	0.0	0.0	6
Spacecraft Contingency	188.2	414.9		104.2	104.2	
Satellite Summary						
EOL Power				**1542.8**	**1542.8**	
BOL Power				**3360.9**		
Dry Mass	**1211.6**	**2671.6**				
Orbit Insertion Propellant	1346.7	2969.5				
On-Orbit Propellant	78.5	173.2				
Pressurant	4.2	9.2				
Wet Mass	**2641.0**	**5823.4**				

Table 3.14. Configuration 12: A Sat D1

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**487.9**	**1075.7**	**40%**	**1069.6**	**1069.6**	
Custom Payload	400.0	882.0		700.0	700.0	5
Payload Communications	70.3	155.0		295.7	295.7	5
Payload Contingency	17.6	38.7		73.9	73.9	
Spacecraft	**910.5**	**2007.6**		**553.6**	**553.6**	
Propulsion	160.8	354.6	14%	0.1	0.1	6
ADACS	63.0	138.9	5%	138.3	138.3	5
TT&C	16.8	37.1	1%	39.1	39.1	9
Command & Data Handling	17.8	39.2	2%	18.7	18.7	6
Thermal	33.8	74.6	3%	229.7	229.7	9
Power	169.3	373.3	14%	0.0	0.0	6
Structure	238.8	526.6	20%	0.0	0.0	6
Spacecraft Contingency	210.1	463.3		127.8	127.8	
Satellite Summary						
EOL Power				**1623.2**	**1623.2**	
BOL Power				**2478.0**		
Dry Mass	**1398.3**	**3083.3**				
Orbit Insertion Propellant	1704.7	3758.8				
On-Orbit Propellant	593.8	1309.4				
Pressurant	5.5	12.2				
Wet Mass	**3702.4**	**8163.8**				

Table 3.15. Configuration 13: B Sat D2

	Mass			Power [W]		NASA TRL
	[kg]	[lbs]	%dry	Daylight	Eclipse	
	Time (min) -->			1366.7	69.4	
Payload	**487.9**	**1075.7**	**42%**	**1069.6**	**1069.6**	
Custom Payload	400.0	882.0		700.0	700.0	5
Payload Communications	70.3	155.0		295.7	295.7	5
Payload Contingency	17.6	38.7		73.9	73.9	
Spacecraft	**835.4**	**1842.1**		**538.9**	**538.9**	
Propulsion	126.3	278.6	11%	0.1	0.1	6
ADACS	61.0	134.5	5%	138.3	138.3	5
TT&C	16.8	37.1	2%	39.1	39.1	9
Command & Data Handling	17.8	39.2	2%	18.7	18.7	6
Thermal	32.1	70.9	3%	218.4	218.4	9
Power	160.9	354.7	14%	0.0	0.0	6
Structure	227.7	502.0	20%	0.0	0.0	6
Spacecraft Contingency	192.8	425.1		124.4	124.4	
Satellite Summary						
EOL Power				**1608.5**	**1608.5**	
BOL Power				**2457.7**		
Dry Mass	**1323.3**	**2917.8**				
Orbit Insertion Propellant	1625.0	3583.1				
On-Orbit Propellant	571.2	1259.5				
Pressurant	5.3	11.7				
Wet Mass	**3524.7**	**7772.0**				

3.3 Configurations Summary

Of the 13 spacecraft configurations generated, the configurations fell into five different spacecraft designs. They were the baseline A Sat, B Sat, C Sat, AB Sat, and MEO Sat. Configuration 12 and 13 are special cases of the baseline A Sat and B Sat spacecraft. These spacecraft have a spacecraft design life of 15 years. They each also carry one less payload than the baseline spacecraft. Their sensor payloads are only designed to operate for 10 years. The concept of operations for these two spacecraft is that they would serve as communication spacecraft for the entire 15 years. Table 3.16 shows a top-level summary of the five primary spacecraft configurations.

Table 3.16. Spacecraft Configurations Summary

	A Sat		B Sat		C Sat		AB Sat		MEO Sat	
	Mass [kg]	Power [W] Daylight	Mass [kg]	Power [W] Daylight	Mass [kg]	Power [W] Daylight	Mass [kg]	Power [W] Daylight	Mass [kg]	Power [W] Daylight
Payload	**537.9**	**1269.6**	**537.9**	**1269.6**	**387.8**	**1043.5**	**998.7**	**2241.2**	**396.0**	**1130.1**
Custom Payload	450.0	900.0	450.0	900.0	314.0	746.1	900.0	1800.0	312.3	661.5
Payload Communications	70.3	295.7	70.3	295.7	59.0	237.9	78.9	353.0	66.9	374.9
Payload Contingency	17.6	73.9	17.6	73.9	14.8	59.5	19.7	88.2	16.7	93.7
Spacecraft	**894.5**	**559.0**	**901.3**	**560.4**	**746.1**	**492.2**	**1640.4**	**837.0**	**659.1**	**421.9**
Propulsion	136.6	0.2	136.6	0.2	115.6	0.1	224.9	0.1	21.9	0.9
ADACS	65.0	138.3	65.0	138.3	53.0	138.3	69.0	138.3	38.0	95.3
TT&C	16.8	39.1	16.8	39.1	16.8	39.1	16.8	39.1	16.8	39.1
Command & Data Handling	11.2	15.3	11.2	15.3	11.2	15.3	11.2	26.7	11.2	15.3
Thermal	34.9	237.1	35.0	238.1	27.3	185.8	87.6	439.7	25.6	173.9
Power	177.5	0.0	181.7	0.0	154.2	0.0	295.9	0.0	209.4	0.0
Structure	246.1	0.0	247.1	0.0	195.7	0.0	556.4	0.0	184.0	0.0
Spacecraft Contingency	206.4	129.0	208.0	129.3	172.2	113.6	378.5	193.1	152.1	97.4
Satellite Summary										
EOL Power		**1828.7**		**1830.0**		**1535.7**		**3078.2**		**1552.0**
BOL Power		**2706.9**		**2723.4**		**2282.4**		**4661.2**		**3378.8**
Dry Mass	**1432.3**		**1439.2**		**1133.9**		**2639.0**		**1055.1**	
Orbit Insertion Propellant	1683.5		1691.0		1244.6		2937.6		0.0	
On-Orbit Propellant	480.3		482.4		315.3		671.3		45.0	
Pressurant	5.4		5.4		4.0		9.5		0.1	
Wet Mass	**3601.6**		**3618.1**		**2697.9**		**6257.4**		**1100.1**	

The spacecraft listed in Table 3.16 will be discussed in greater detail in each of the subsystem sections of this report.

3.4 Common Spacecraft Configuration Observations

For all of the spacecraft configurations, there were no additional mass and power margins placed on the payloads at the customer's direction with the exception of the communications payload. Mass and power margins were carried for the spacecraft bus and communications payload, which was 30% and 25%, respectively.

All of the spacecraft were designed to be injected into a transfer orbit. At that transfer orbit, the on-board propulsion system would circularize the spacecraft orbit to either GEO or MEO. All of the spacecraft were launched by either the Delta or Atlas EELV. All of the spacecraft configurations, with one exception, were dual manifested with another spacecraft. Table 3.17 shows the launch vehicle used along with the launch vehicle margin for the spacecraft configurations.

It is desirable to have a launch margin that is greater than approximately 10%. Two of the launch configurations only have one-half of the desired margin.

Table 3.17. Spacecraft Configurations Launch Vehicle Manifest

Satellite Configurations	# per LV	Launch Vehicle	Launch Mass Margin
Baseline A Sat Baseline B Sat	1 1	Atlas V 551	9%
A Sat minus Comm B Sat minus Comm	1 1	Atlas V 531	5%
Baseline A Sat B Sat MIT	1 1	Atlas V 551	5%
Baseline C Sat A Sat minus Comm	1 1	Atlas V 531	11%
C Sat minus Comm Baseline A Sat	1 1	Atlas V 531	9%
Baseline AB Sat	1	Atlas V 531	12%
AB Sat minus Comm	2	Delta IV Heavy	8%
Baseline MEO Sat	4	Delta IV Heavy	25%
MEO Sat Transfer A Sat minus Comm	1 1	Atlas V 531	12%
A Sat D1 B Sat D2	1 1	Atlas V 551	8%

4. Configuration

Scott Szogas

4.1 Overview

As a normal output, the CDC generates a conceptual-level satellite configuration (three-dimensional CAD model) in order to provide geometric information about the satellite being studied. The configuration is built up from the primary building blocks of the satellite, including the spacecraft bus, solar arrays (sized accordingly), and payload suite (sensors and antennas). The structure subsystem expert, using historical satellite data and the overall satellite mass, determines the proper volume for the spacecraft bus. The dimensions of the satellite are then chosen to provide the proper satellite bus volume. The sensors are constructed based on customer-provided drawings or sketches. The communications antenna dimensions and configuration are provided by the communications subsystem expert and are based on the mission requirements and concept of operations.

The spacecraft configuration model is constructed to provide a geometric baseline for the satellite. The sizes of the array and bus structure are coordinated with the power and structure subsystems during the study. The geometric model is also used to generate moments of inertia of the satellite based on the locations of the various components being modeled. The attitude determination and control subsystem expert uses the mass moment of inertia to determine the capability of the actuators required to control the satellite attitude. Radiator areas and locations are also discussed with the thermal expert to assure that adequate thermal dissipation is present. Finally, a stowed configuration is constructed to assure that the satellite can be stowed within the launch vehicle fairing. For this study, multiple launch manifests were required. There are several methods available to accommodate stacking satellites for multiple launch manifests: pancake style, by beefing up the existing satellite structure (i.e., DSCS, StackSat); cake-slice style by creating a tall-thin satellite configuration and radial stacking (e.g., Iridium); or can-type style by designing an external payload adapter (i.e., Delta DPAF). The later was chosen as the most mass efficient for the GOES-R configurations. A more detailed analysis would be necessary to validate this assumption.

4.2 Analysis

The deployed on-orbit configurations of the 10 principal spacecraft generated during the study sessions are presented in Figures 4.1 through 4.13. In general, the payload complement presented in these configurations was accommodated without much difficulty. The baseline AB Sat configuration was the most stressing as far as utilization of the nadir panel mounting surface, and required that the nadir panel be placed in a vertical configuration during launch. A single-axis articulated solar array was chosen to provide an unobstructed field of view for the sensor radiators. An in-depth FOV analysis should be performed for the configurations in order to verify that the sensors and antenna all have adequate clear field-of-regard.

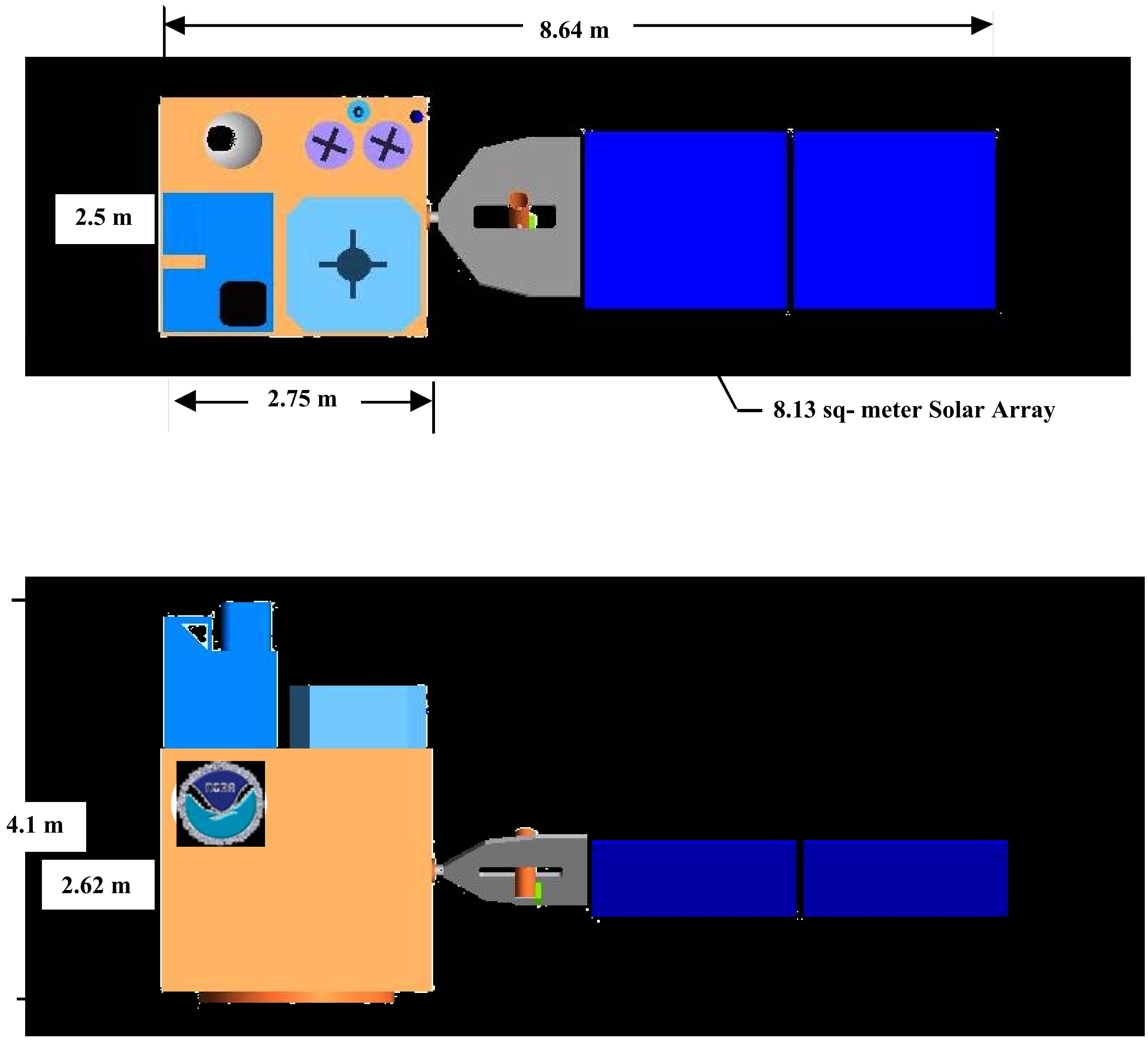

Figure 4.1. Baseline A Sat configuration with key dimensions.

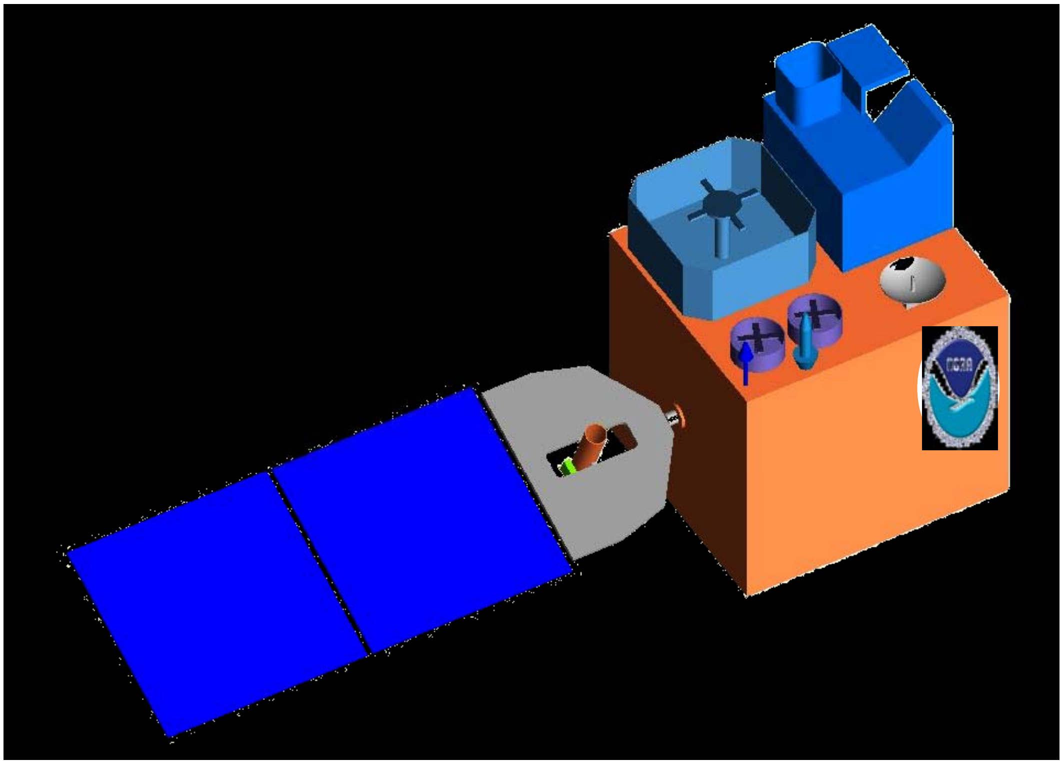

Figure 4.2. Baseline A Sat configuration.

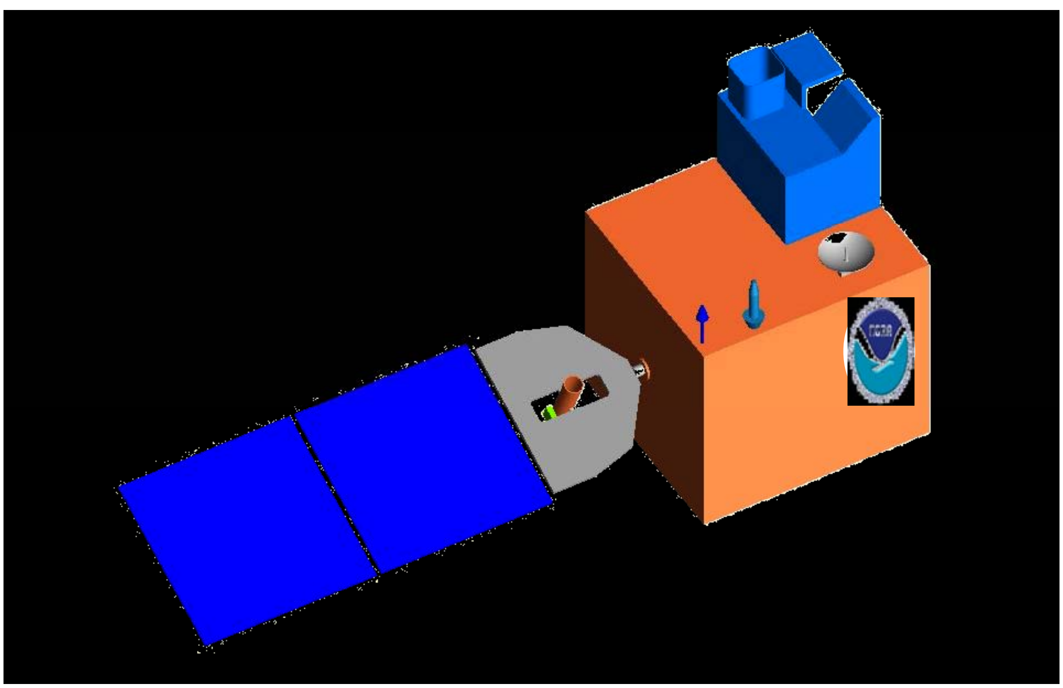

Figure 4.3. Baseline A Sat minus communications configuration.

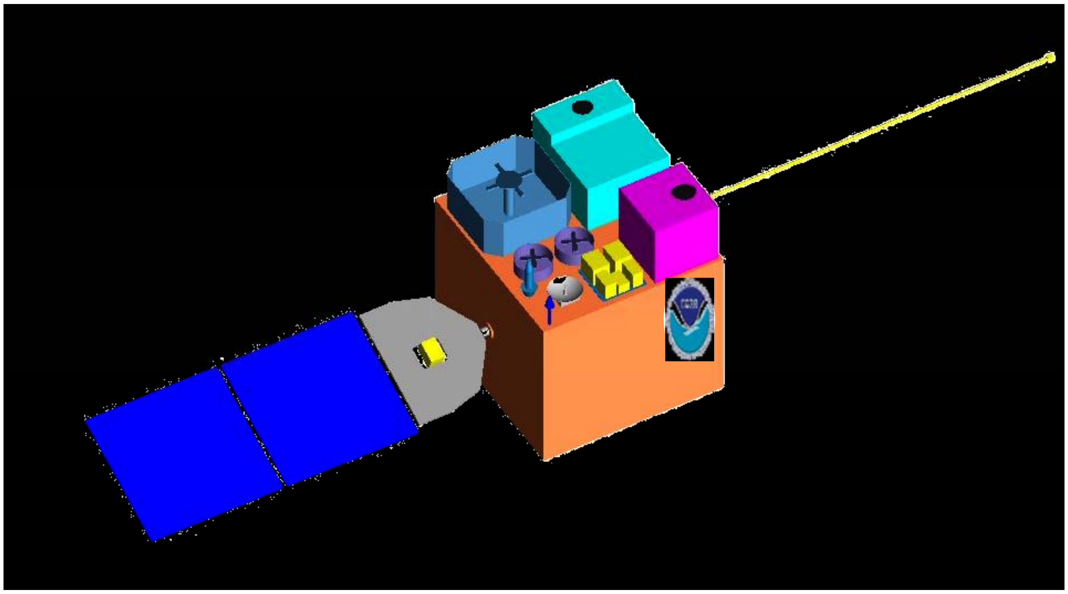

Figure 4.4. Baseline B Sat configuration.

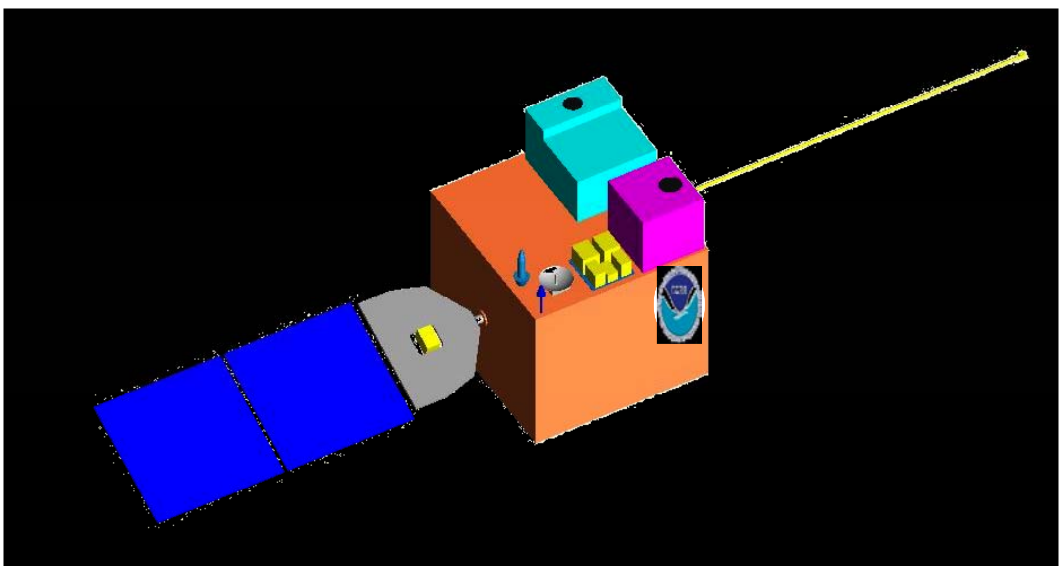

Figure 4.5. Baseline B Sat minus communications configuration.

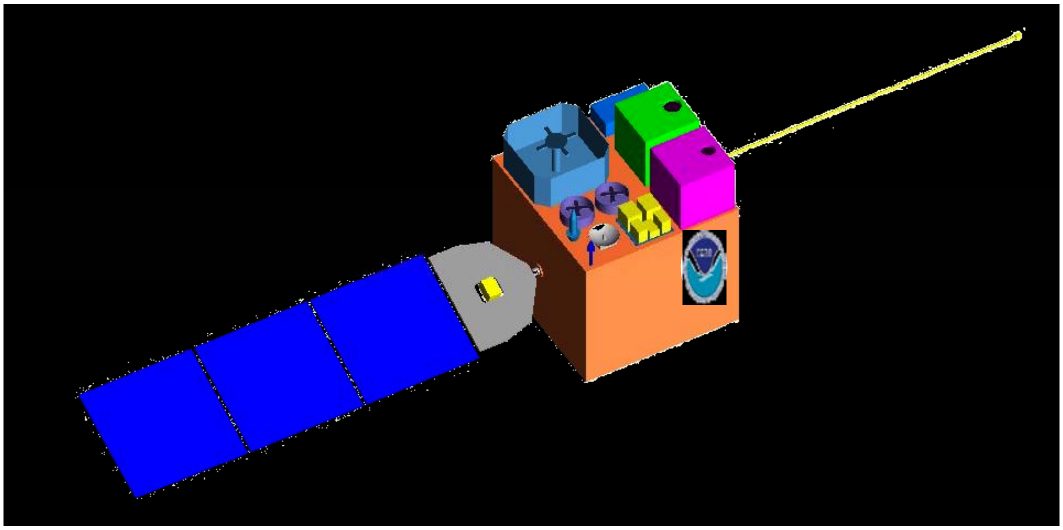
Figure 4.6. B Sat MIT configuration.

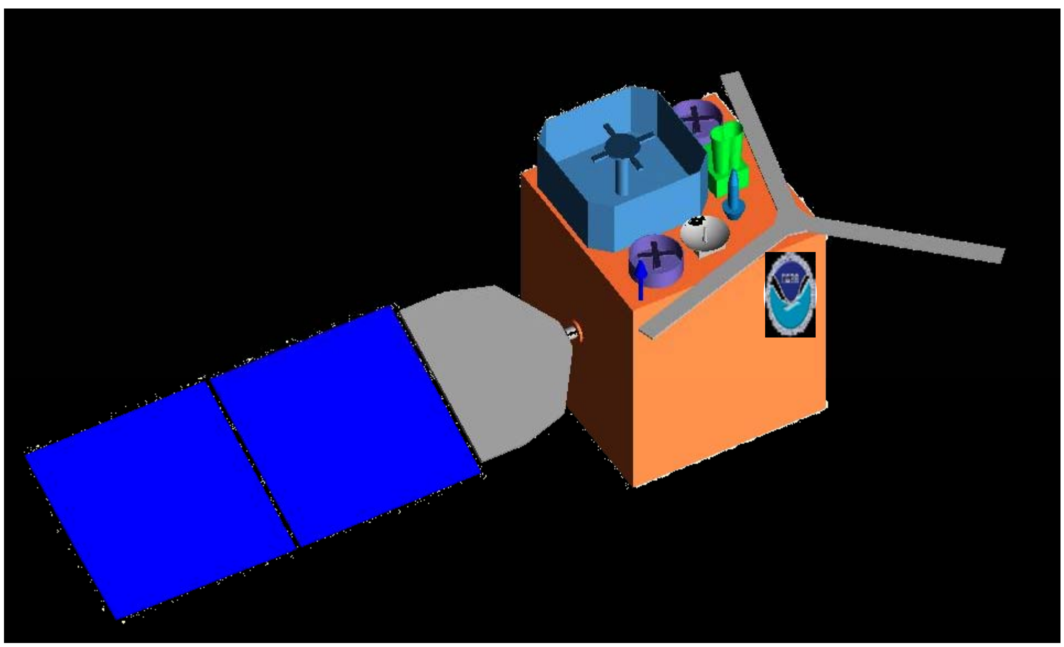
Figure 4.7. Baseline C Sat configuration.

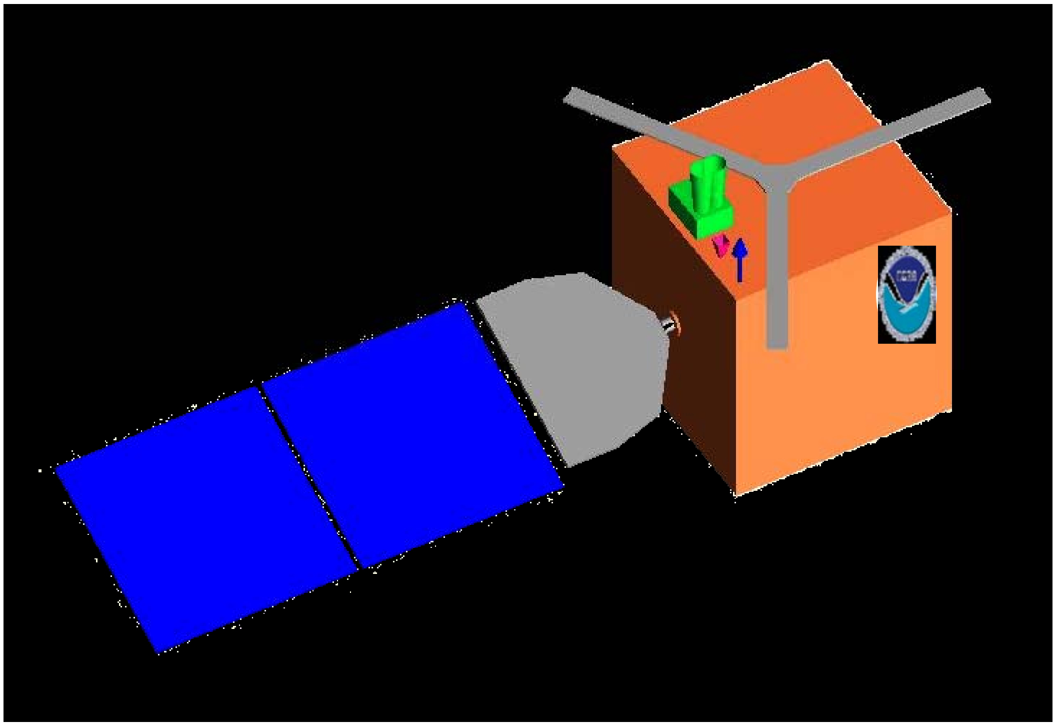

Figure 4.8. Baseline C Sat minus communications configuration.

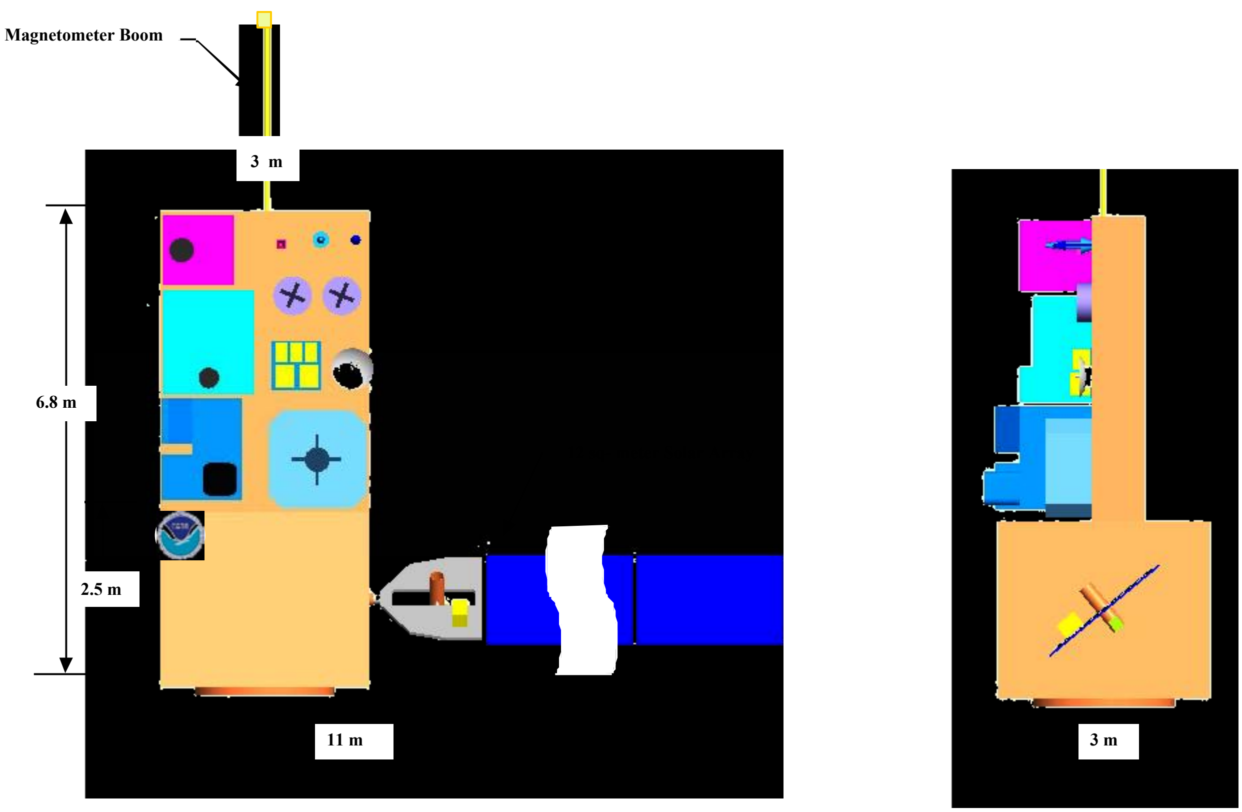

Figure 4.9. Baseline AB Sat configuration with key dimensions.

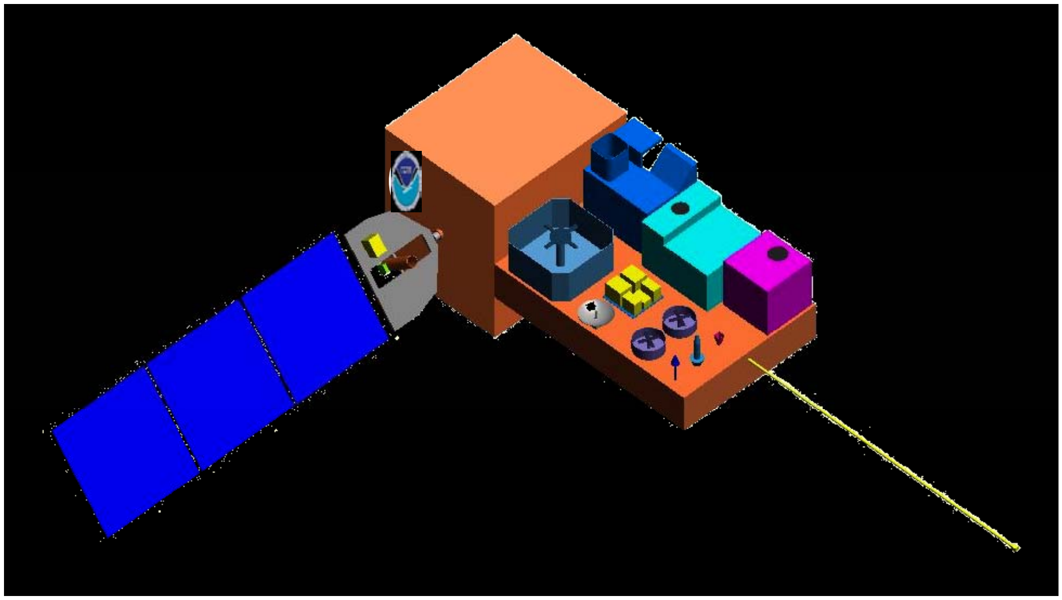

Figure 4.10. Baseline AB Sat configuration.

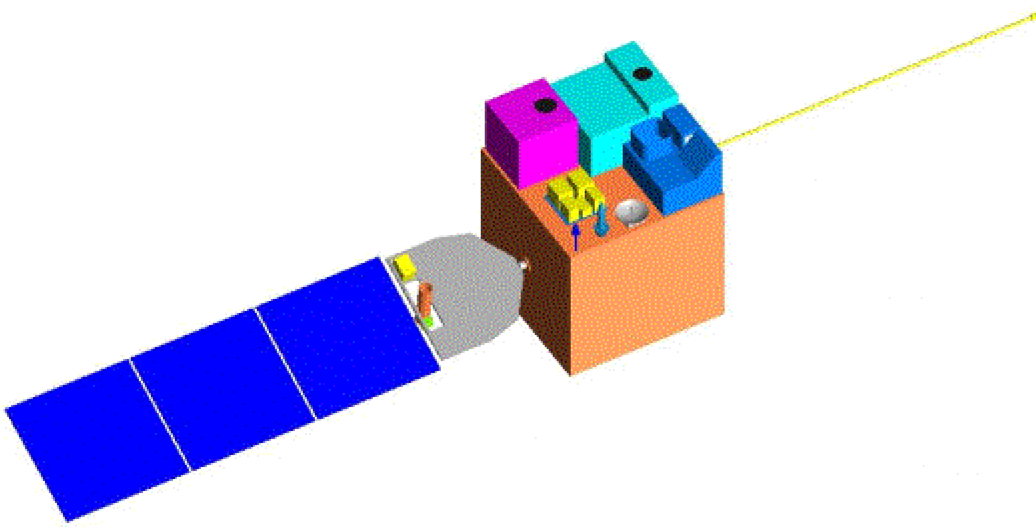

Figure 4.11. Baseline AB Sat minus communications configuration.

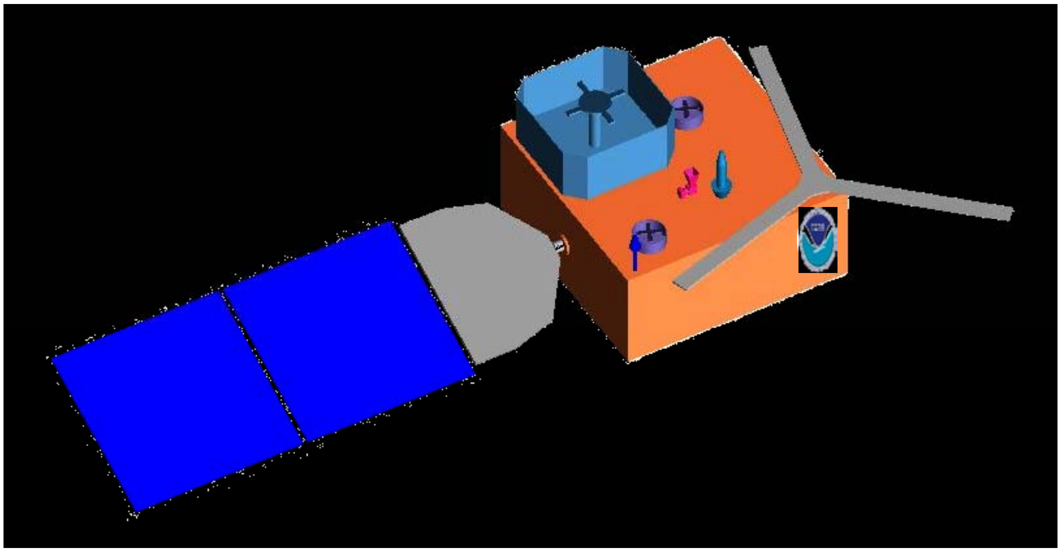

Figure 4.12. Baseline MEO Sat configuration.

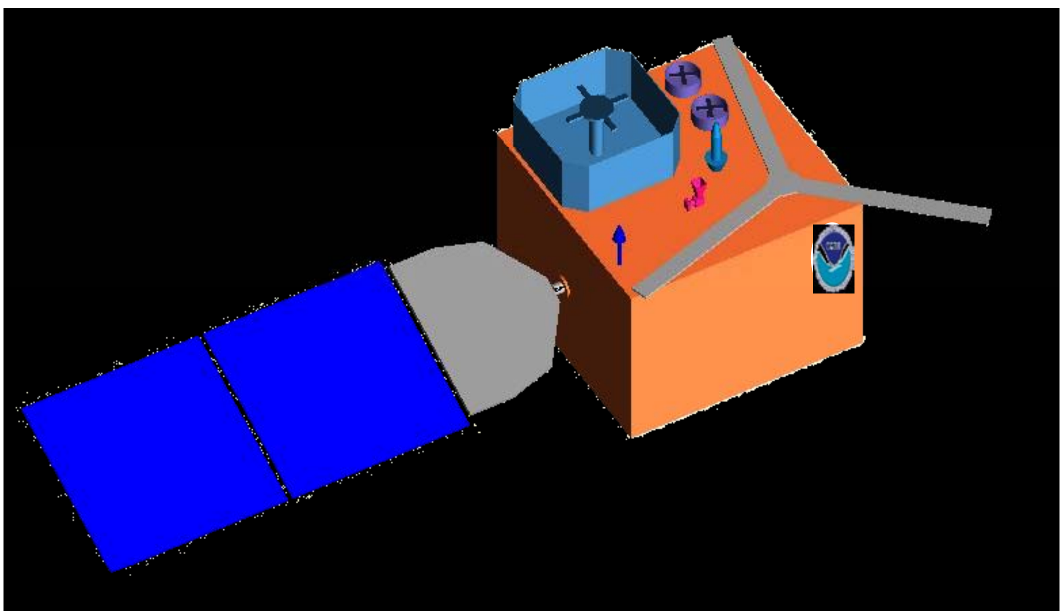

Figure 4.13. Baseline MEO Sat GEO to MEO insertion configuration.

Figures 4.14 through 4.22 show notional launch configurations for the various GOES spacecraft architectures. Although there are several options available for stacking satellites, a payload adapter is shown to stack the satellites for multiple launch manifest missions. This approach was selected as the lowest mass method of stacking satellites, but would limit the stowed diameter of the lower satellite to 4 m. All launch configurations are shown within 5 m (Delta IV or longer Atlas-V).

Figure 4.14. Baseline A Sat and Baseline B Sat stowed.

Figure 4.15. Baseline A Sat minus communications and baseline B Sat minus communications stowed.

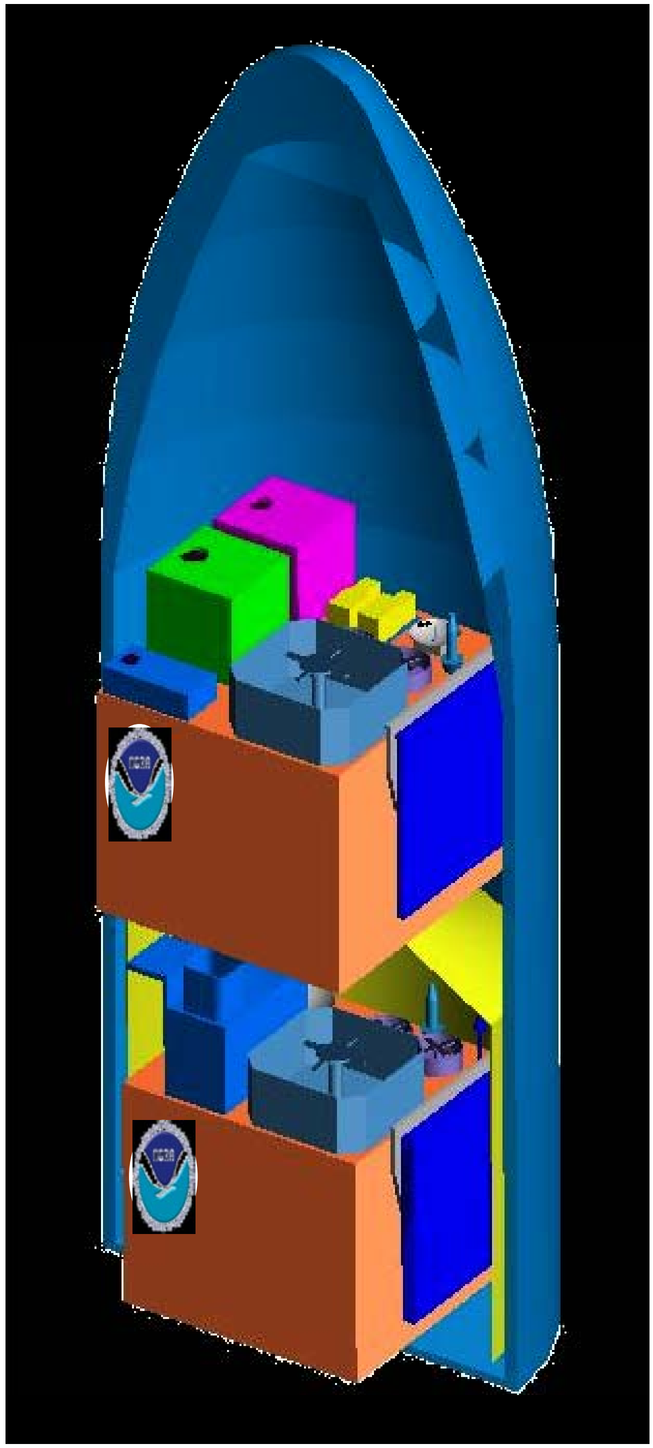

Figure 4.16. Baseline A Sat and B Sat MIT stowed.

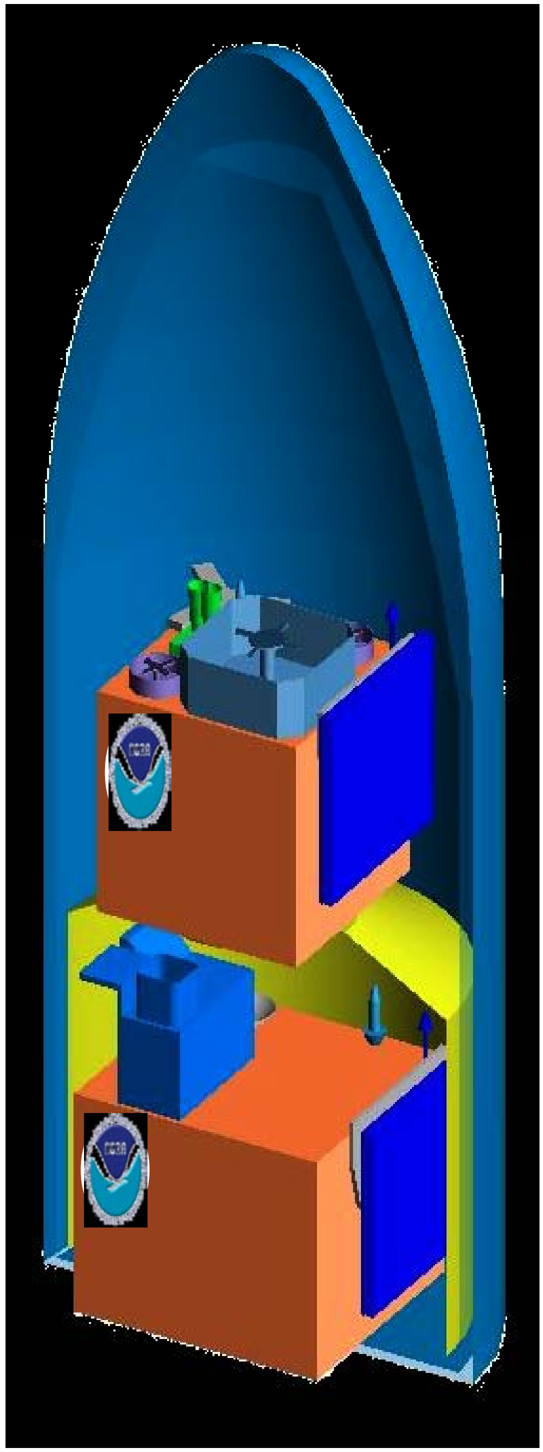

Figure 4.17. Baseline A Sat minus communications and baseline C Sat stowed.

Figure 4.18. Baseline A Sat and baseline C Sat minus communications stowed.

Figure 4.19. Baseline AB Sat stowed.

Figure 4.20. Baseline AB Sat minus communications stowed.

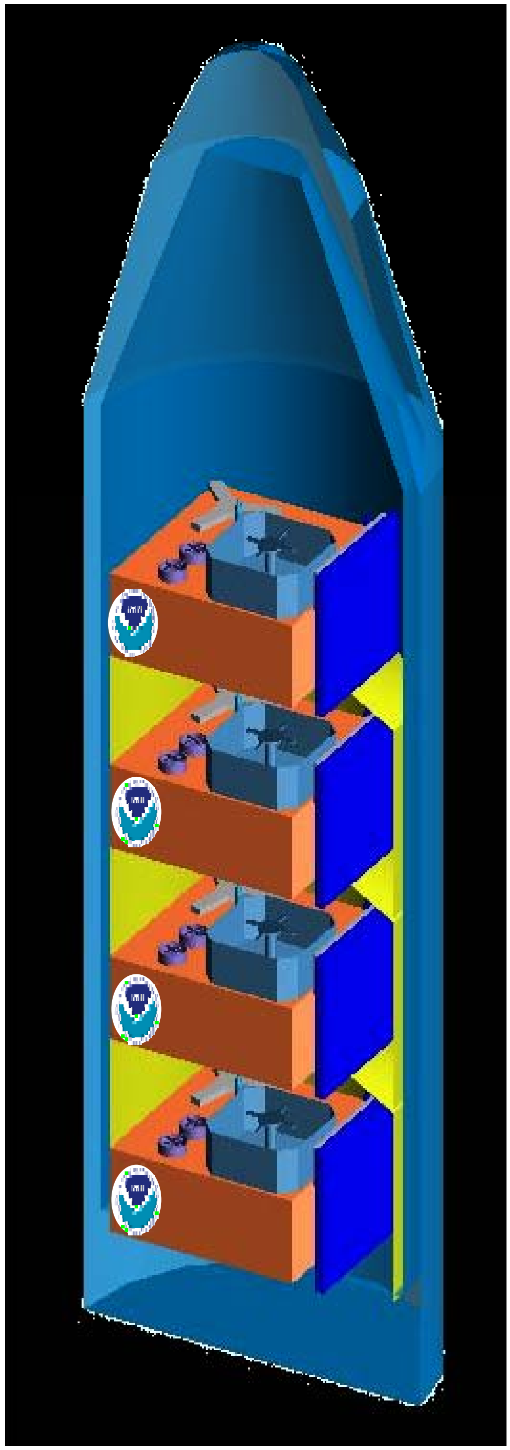

Figure 4.21. Baseline MEO Sat stowed.

Figure 4.22. Baseline A Sat minus communications and baseline MEO Sat GEO to MEO insertion stowed.

This page intentionally blank.

5. Astrodynamics

Tom Lang and Laura Speckman

5.1 Overview

There were two basic spacecraft architectures from the standpoint of astrodynamics: (1) GEO satellite architectures, and (2) MEO satellite architectures.

5.2 GEO Satellite Architectures

A number of GEO configurations were studied, but all were identical from the standpoint of astrodynamics. As in the previous GOES-R study, the total constellation consists of six spacecraft in a GEO.

North/South (N/S) and East/West (E/W) station-keeping requirements were sized for 9 years since the spacecraft is stored for 1 year on the ground. To hold a GEO satellite at 0.5° inclination requires 50 m/s per year, so N/S station-keeping delta-V is (50 m/s per year times nine years) 450 m/s. East/West station-keeping for GEO is 2 m/s per year, so E/W station-keeping is (2 m/s times 9 years) 18 m/s. There is no drag makeup delta-V at GEO.

A total of 4 repositions at 3° per day per satellite were requested by the customer, for a total delta-V of 68.2 m/s. At satellite end-of-life (EOL), disposal to an orbit +300 km above GEO requires a delta-V of 10.9 m/s.

The satellites will experience a maximum eclipse duration of 69.4 min, and there will be approximately 92 eclipses per year.

5.3 MEO Satellite Architecture

In this architecture, 4 MEO satellites were evenly spaced in an equatorial orbit at an altitude of 10,385 km. North/South station-keeping delta-V was 0 m/s for the MEO architecture. This is typical for MEO constellations, as there is usually no N/S station-keeping requirement for MEO spacecraft. East/West station-keeping is 1 m/s per year, so E/W delta-V is (1 m/s times 12 years) 12 m/s. There is no drag makeup delta-V at MEO. A total of 2 repositions at 3° per day required a total of 18 m/s of maneuver delta-V. At satellite EOL, disposal to +200 km above MEO required a delta-V of 28.8 m/s.

Two possible values of inclination for the MEO were examined, 0° and 180°. The station-keeping and eclipsing requirements are the same for both orbits, but the payload mass that can be carried to the two inclinations can be very different. Some details of the equatorial MEO are shown in Figures 5.1 through 5.3.

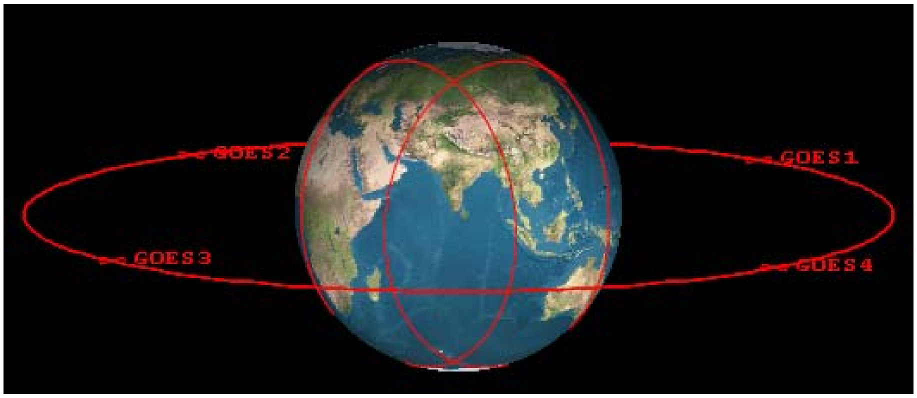

Figure 5.1. GOES four-satellite MEO architecture.

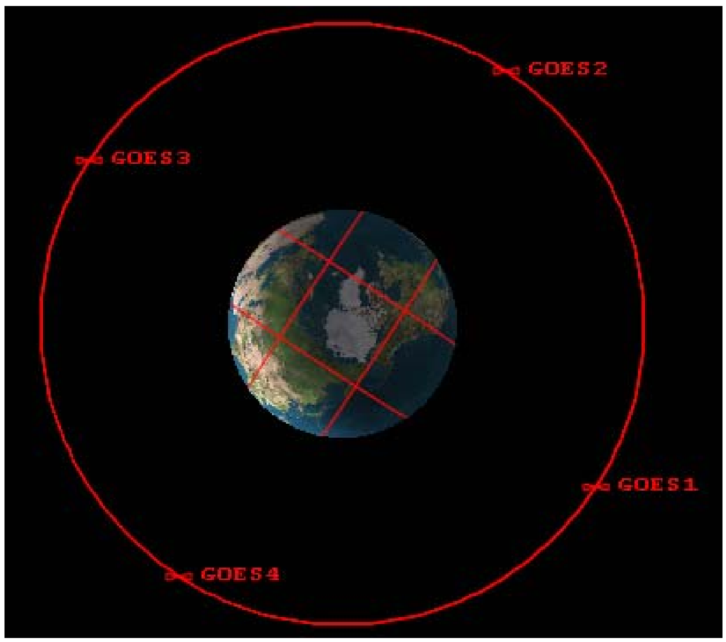

Figure 5.2. GOES four-satellite MEO architecture, polar view.

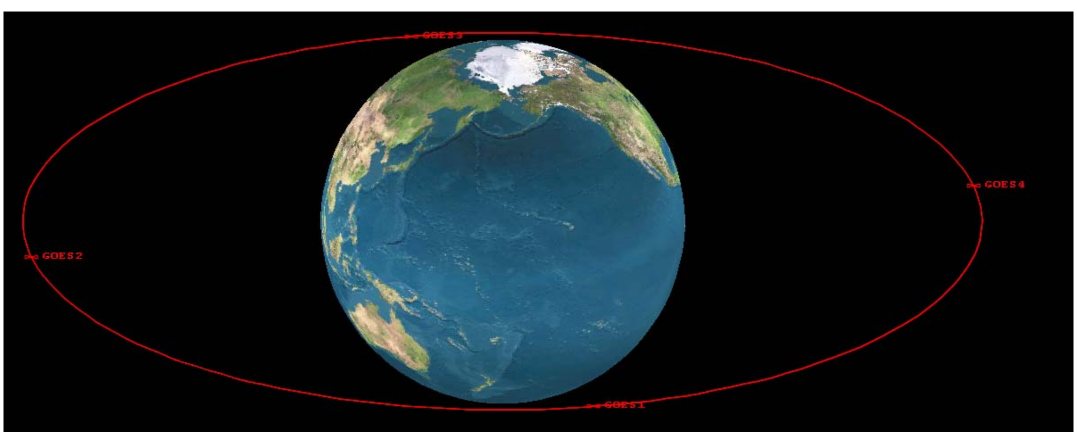

Figure 5.3. View from the sun at summer solstice.

Figure 5.1 shows the GOES four-satellite MEO architecture. The sensors' FOVs are shown projected on the Earth's surface.

Figure 5.3, which is a view of the MEO from the sun at summer solstice, shows the situation in which the spacecraft orbit is mostly sunlit. This view is what the sun sees and it shows that the orbit barely escapes being eclipsed. However, if an allowance for the Earth's atmosphere is made, it is possible that the orbit is eclipsed. Eclipse is also possible with N-S drift. This means that the equatorial MEO is always eclipsed by the Earth 4 times per day for a total of 1,461 eclipses per year. The longest of these eclipses is 44 min.

Also note that the views are the same whether the constellation is at an inclination of 0° or 180°. The inclination only affects the direction in which the satellites are rotating (0° = Eastward; 180° = Westward). Table 5.1 shows some information on the retrograde (inclination = 180°) orbit. The information is the same for the posigrade (inclination = 0°) orbit, except that the Earth's relative period (time to come back to the same point on the equator) is eight hours.

Table 5.1. Data on the Retrograde MEO Orbit

Period	360 min (4 h 49 min Earth relative)
Altitude	10385 km
Inclination	180° (retrograde)
Delta Velocities	
N/S	0 m/s
E/W	12 m/s (1 m/s/yr*12yr)
Repos	18 m/s (2@3°/day
EOL Disposal	28 m/s (+200km)
Eclipse	
Max duration	44 min
Max frequency	1461/yr (4 times per day)

5.4 MEO Dual Launch

To enable a dual launch, where one satellite is delivered to GEO and the other to MEO would require some additional delta-V. Three cases of interest were identified.

(1) Go from GTO (167 x 35,786, I = 27) to MEO (10,385 x 10,385, I = 0)

(2) Go from GEO (35,786 x 35,786, I = 0) to MEO (10,385 x 10,385, I = 0)

(3) Go from GTO (167 x 35,786, I = 27) to MEO (10,385 x 10,385, I = 180)

The GEO-to-MEO case was used because it required the least amount of delta-V. Table 5.2 shows the delta-Vs required for each of the transfers.

Table 5.2. Delta-Vs Required for Dual-Launch Orbit Transfers

Case	Orbit (km)	Inclination (°)	Burn Altitude (km)	Plane Change (°)	Delta-V (m/s)
1	167 x 35,786	27	35,786	24	1,079
	10,385 x 35,786	3	10,385	3	997
2	10,385 x 35,786	0	35,786	0	756
	10,385 x 10,385	0	10,385	0	957
3	167 x 35,786	27	35,786	162	3,867
	10,385 x 35,786	179	10,385	1	962

6. Command and Data Handling

Ron Selden

6.1 Overview

The Command and Data Handling (C&DH) subsystem is fully redundant due to the ten-year length of the mission. The C&DH subsystem consists of two processors, two input/output controller boards, two solid-state mass memory devices, two power supplies, and a chassis. All of the components are redundant. Figure 6.1 presents a block diagram of the C&DH model.

The C&DH architecture did not change between the configurations in the baseline studies. The architecture did change for the excursion studies, A Sat D1 and B Sat D2. On these two spacecraft, the architecture changes to accommodate the extended lifetime of 15 years by adding an additional processor. A new purpose-built board is added to select between the two input/output boards and the three processors. A different purpose-built board is also added to select among the three processors and the three solid-state memory boards.

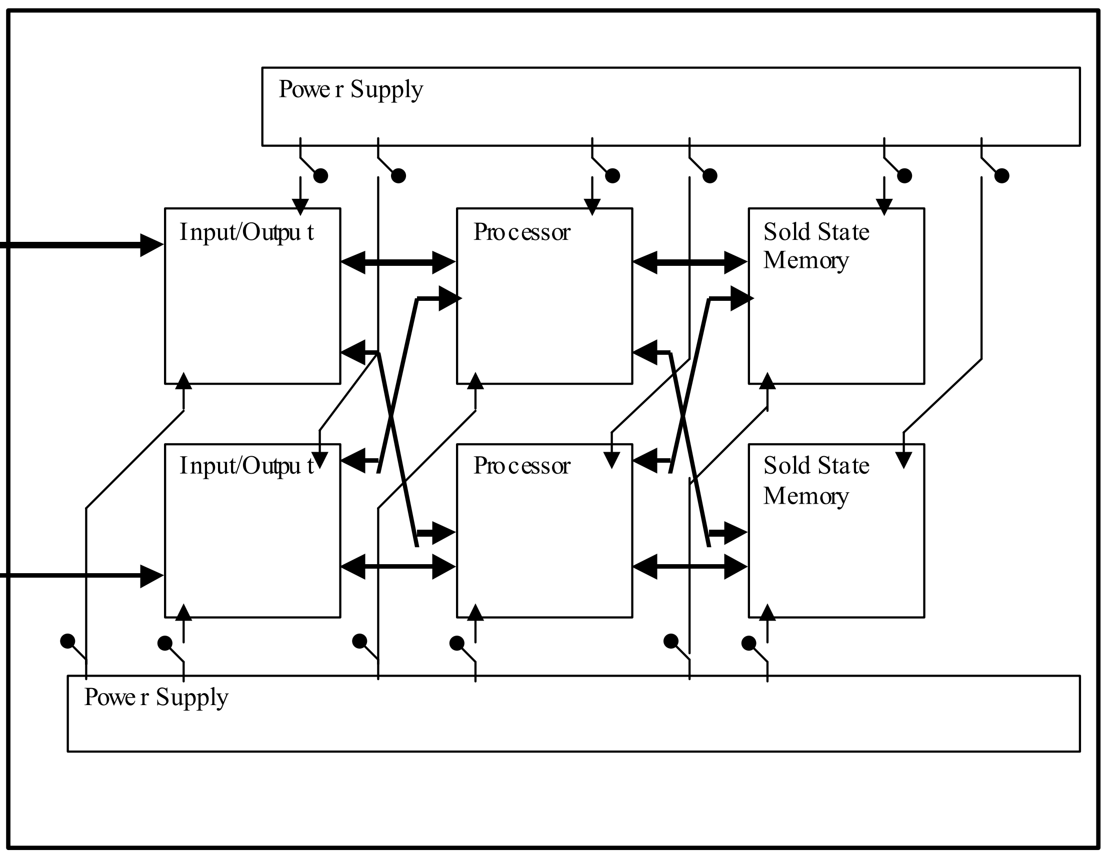

Figure 6.1. C&DH model block diagram.

Table 6.1 presents the details of the C&DH subsystem design and is the same for the following spacecraft configurations with the exceptions of the baseline AB Sat and baseline AB Sat minus communications.

Table 6.2 presents the details of the C&DH subsystem design for the following spacecraft configurations: baseline AB Sat and baseline AB Sat minus communications. The hardware for all spacecraft configurations is the same, but the baseline AB Sat and baseline AB Sat minus communications have increased power. The power increases because the processor operates at a higher speed to accommodate the additional payloads and associated throughput.

Table 6.3 shows the mass and power summary for the A Sat D1 and B Sat D2 spacecraft.

Table 6.1. C&DH Subsystem Design Summary All Configurations Except Baseline AB Sat and Baseline AB Sat Minus Communications

	Units	Mass (kg)	Power (W)	NASA TRL
Command & Data Handling		**11.2**	**15.3**	**6**
Processor	2	1.2	6.0	7
Input / Output Controller	2	1.0	2.5	5
Solid-state Memory	2	5.0	3.0	6
Power supply	2	2.0	3.8	7
Chassis	1	2.0	0.0	7

Table 6.2. C&DH Subsystem Design Summary for Baseline AB Sat and Baseline AB Sat Minus Communications

	Units	Mass (kg)	Power (W)	NASA TRL
Command & Data Handling		**11.2**	**26.7**	**6**
Processor	2	1.2	12.0	7
Input / Output Controller	2	1.0	5.0	5
Solid-state Memory	2	5.0	3.0	6
Power supply	2	2.0	6.7	7
Chassis	1	2.0	0.0	7

Table 6.3. C&DH Subsystem Design Summary for A Sat D1 and B Sat D2

	Units	Mass (kg)	Power (W)	NASA TRL
Command & Data Handling		**17.8**	**18.7**	**6**
Processor	2	1.8	6.0	7
Input / Output Controller	2	1.0	2.5	5
Cross Bar Interface Unit	3	1.5	2.5	5
Solid-state Memory	2	7.5	3.0	6
Power supply	3	3.0	4.7	7
Chassis	1	3.0	0.0	7

6.2 Technology Assumptions

There are no special technology developments taken into account for this study. The processor selected is the most advanced space-qualified hardware currently available. The power and mass numbers are based on current technology, and no dramatic improvements are foreseen by the technology freeze date of 2008. The technology freeze model assumes a 10% per year reduction in mass and power. These assumptions are incorporated into the results presented here. No new space processors are currently in development. Because the model assumes a reduction in mass and power for later technology freeze dates, the mass and power are about half of today's (2003) available hardware. This is reflected in the mass and power fractions being at the low end of the typical range.

The high-speed input/output board that interfaces to the payload equipment is a new design and may require significant development efforts. The high-speed payload interface is envisioned to be a Firewire or equivalent connection. A detailed trade study must be completed to determine the specific hardware design and approach for this interface. This board is purpose-built hardware and will incorporate the latest technology available as of the technology freeze date.

The capacity of the mass memory is a high estimate at slightly more than 100 MB. Mass memory and power are not a significant component of the spacecraft total. This is a conservative estimate. An in-depth look at the required capacity needs to be completed, but this will not significantly affect the total mass and cost. The estimate used for this study should be considered as an upper bound for the capacity of the mass memory.

6.3 Component Descriptions

6.3.1 Power supply: Estimate

- Mass: 1 kg per board times two boards equals 2 kg
- Efficiency: 75%
- Power: 3 W
- Mass: 2 kg

6.3.2 Solid-state memory (Mass Memory)

- Capacity: 1 Gb
- Mass: 2.5 kg per board times two boards equals 5 kg
- Power: 3 W

6.3.3 Processor: Rad-Hard Power PC (RHPPC)

- Throughput: ~200 MIPS de-rated to one half speed to save power
- Mass: 0.8 kg per board times two boards equals 1.2 kg
- Power: 12 W at full speed de-rated to 6 W

6.3.4 Input/output controller: Custom design

- Mass: 0.5 kg per board times two boards equals 1 kg
- Power: 2.5 W

6.3.5 Chassis

- Mass: 2 kg

This page intentionally blank.

7. Telemetry, Tracking, and Command

John O'Donnell

7.1 Overview

The telemetry, tracking, and command (TT&C) subsystem is configured to evaluate the design of the command and control radio frequency (rf) link in support of the TT&C subsystem on the GOES-R spacecraft. Commands originate from a continental US-based control facility and are transmitted to the spacecraft by a network of remote ground facilities. Vehicle telemetry is received at the control facility by the network's return link.

7.2 Design Summary

The TT&C subsystem is designed and sized to support the standard NASA unified S-band link for activities that include launch, early orbit checkout, transfer orbit (if applicable), and operational orbit ranging/health status and anomaly resolution. The TT&C subsystem design assumptions are presented in Table 7.1.

The TT&C subsystem unit design is presented in Table 7.2. The mass and power estimates represent totals for the unit quantities, which provide a fully redundant system.

Table 7.1. Design Assumptions per Module/Link Capability

	TT&C Uplink	TT&C Downlink
Frequencies	Unified S-band command link frequency band	Unified S-band telemetry link (2200 to 2300 MHz range)
Data Rates	Command = 2 kb/s	Tlm = 8 kb/s during launch, early orbit Tlm = 32 kb/s during nominal on-station operations
Antennas	RTS antenna = minimum 10 m dia	Spacecraft: Omni (qty = 2) EC Horn (qty = 1)

Table 7.2. TT&C Subsystem Design

	Unit Qty	Total Mass (lb)	Total Power (W)	NASA TRL
TT&C Subsystem (SGLS Link)				
Omni Antenna	2	0.8	0	9
Horn Antenna Assembly	1	5.0	0	9
Transponder	2	14	32	8
Cmd Signal Conditioning Unit	2	3.6	1.5	8
TLM Base Band Assembly Unit	2	3.6	1.5	8
Local Oscillator	2	3.0	2	8
Comsec	2	2.0	2.5	9
Miscellaneous RF Hardware	1	5.0	0	9
TT&C Subsystem Total:		**37**	**39.5**	

The TT&C subsystem is a flight-proven system, thus the high TRL numbers in Table 7.2. All configurations studied for the GOES-R spacecraft resulted in the same TT&C subsystem design, thus Table 7.2 represents all configurations.

The downlink telemetry rates of 8 kb/s and 32 kb/s are typical, non-stressing rates for a flight telemetry system. It is safe to anticipate that higher rate vehicle/payload telemetry could be transmitted "in-band" in the payload sensor data downlink to the Wallops ground site from the mission orbit. The unified S-band link through the ground control network would be available for scheduled ranging and states of health contacts and anomaly resolution contacts. During launch and orbit transfer, the TT&C system would operate through the two hemispherical coverage patch antennas providing the vehicle near-2π sr coverage. During this time, the data rate is anticipated to operate at 8 kb/s.

Once on orbit at GEO, the TT&C system would switch to an Earth coverage horn antenna providing sufficient gain for increasing the telemetry rate to 32 kb/s. The link requires a 10-W rf solid-state power amplifier (SSPA) transmitter for link closure during all mission orbit phases. At the time of the GOES-R satellite development, a SGLS-USB dual-mode transponder will be an available off-the-shelf item providing ranging turnaround, thus allowing for command and control compatibility between NASA ground network and the Air Force Satellite Control Network of ground stations. The command signal conditioning unit (SCU) provides the command decoder function and is the forward link interface to the C&DH subsystem. The telemetry baseband assembly unit (BBAU) provides the associated functional interface between the C&DH subsystem and telemetry return link for telemetry encoding.

Both the SCU and the BBAU are anticipated to be integrated slices within the C&DH. The comsec unit is anticipated to be the L3Comm MFU flight-qualified unit, providing cardholder command decryption/authentication and Pegasus telemetry encryption.

8. Payload Communications

John O'Donnell

8.1 Overview

For each of the configurations studied, the GOES-R payload communication model is designed to satisfy the data requirements of the payloads as proposed by the customer and provide a means of evaluating the overall delta impact to the spacecraft design. Although this summary write-up provides an allocation of mass and powers at the individual communication subsystem levels, the actual payload communication model used during the study evaluated the communication design at a unit level mass and power breakdown, and these details could be made available.

The payload communications subsystem provides the capability to transmit raw sensor data directly to the ground station located at Wallops Island, receive the processed mission data uplink from Wallops Island, and transpond that uplink via a broadcast mode to all in-view ground users. The concept behind this study was to have up to three spacecraft occupying common orbital slots at 75° West and 137° West. Each spacecraft would have a raw sensor data downlink transmission to Wallops Island; yet, only one spacecraft would actively provide the GRB of processed mission data. Before examining the design of the payload communications subsystem for each of the study configurations, an overview of the communication links and their options will be provided.

- Sensor Data Downlink. This communication link is a point-to-point rf link from each of the GOES-R satellites to the Wallops Island ground facility. It provides continuous connectivity of raw sensor data collected by the satellite for processing on the ground. A single frequency band was examined for this link: X-band (8,215–8,400 MHz). Based on the available spectrum and the required downlink data rates, all three spacecraft would frequency share the X-band spectrum band. The modulation of this data link was assumed to be O-QPSK with 15/16 Turbo code forward error correction (FEC) applied.

- GRB. This communication link provides processed data to the user community. It is a transponded link from Wallops Island (uplink) through the GOES-R satellite and globally broadcast to the user community (downlink). A single frequency band combination was examined for this transponded link: X-band uplink (7,190–7,235 MHz) and L-band downlink broadcast (1,683–1,695 MHz). This link operates at 10 Mb/s and employs O-QPSK modulation and 15/16 Turbo code forward error correction. As mentioned, only one of the three spacecraft in each orbital slot would provide this GRB.

- Auxiliary Signal Broadcast. This communication link provides three low-rate, frequency-multiplexed auxiliary data transmissions to the user community. Like the GRB transmission, these multiplexed signals are transmitted from Wallops Island and are transponded by the satellite for user broadcast reception. All configurations occupied the uplink/downlink spectrum of 7,190–7,235 MHz/1,695–1,698 MHz. The three-signal set consists of:

- LRIT: 600 kb/s with QPSK modulation
- EMWIN: 8 kb/s with BPSK modulation
- DCPR: 233 channels of 100, 300, and 1,200 b/s

The spacecraft providing the GRB transmission would also simultaneously provide this auxiliary signal broadcast transmission.

8.2 Design Summary

This subsection summarizes the payload communication designs for all of the spacecraft configurations. All of the design information presented represents the design of the communication system without mass and power contingency included. Within the systems module, a 25% contingency is added to the communication system's mass and power estimates. The spacecraft bus is then sized based on these estimates that include contingency. The estimates provided herein are without the 25% contingency applied.

8.2.1 Configuration 1: Baseline A Sat

The sensors on the baseline A Sat included an ABI sensor producing 60 Mb/s of data and a SXI sensor producing 2.8 Mb/s of data. To support this sensor suite, the communication system included a 63-Mb/s sensor data downlink transmission, a 10-Mb/s GRB transponded transmission, and an auxiliary signal transponded transmission. This configuration involved the following communication system design characteristics:

- Sensor Data Downlink. X-band downlink at 63 Mb/s. This direct downlink to Wallops was designed as a single-polarization transmission using O-QPSK modulation with 15/16 Turbo Code FEC. Detailed link analysis defined a transmit power of nine W rf linear (14 W rf saturated) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. For compatibility with the design of the baseline B Sat communication system, a 10.0-W rf linear (16-W rf saturated) SSPA was used. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The GRB uplink signal is received through the same 0.5-m gimbal antenna used for the sensor data downlink to Wallops. The GRB link analysis showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 14 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of backoff from saturation (36 W rf) was required, when using O-QPSK modulation, for operation of the traveling-wave tube amplifier (TWTA).

- Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set (LRIT, EMWIN, and DCPR) is transponded by the GOES-R satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point. The mass and power included in the communication system accounts for the SSPA required for transmission of the LRIT, EMWIN, and DCPR signals. The associated transceivers are assumed to be included within the payload mass and power allocations.

A conceptual overview of this communication configuration is provided in Figure 8.1.

Table 8.1 summarizes, in a top-level breakdown, the mass and power of the communication subsystem.

As noted, the L-band global broadcast required 14 W of rf power operating in the linear region. For operation of the TWTA, recent analysis indicates that a level of 4.1 dB of backoff is required for O-QPSK modulation. For TWTA rf to DC conversion, a 45% of tube efficiency was assumed. The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

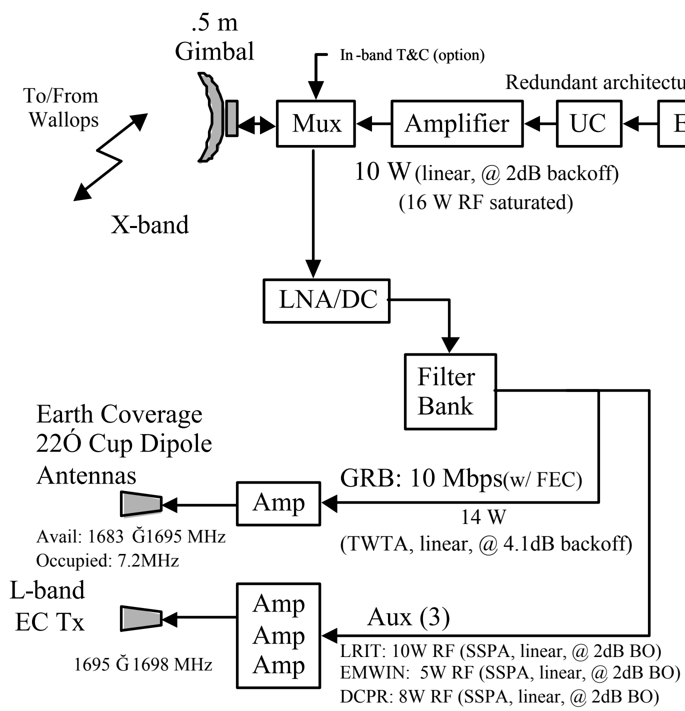

Figure 8.1. Baseline A spacecraft communications system block diagram.

Table 8.1. Baseline A Sat Communication Summary

Baseline A Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink RF Hardware	X-band	16.7	60
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22.0
Broadcast System RF Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	208
	Communication System Total:	**155**	**296**

8.2.2 Configuration 2: Baseline A Sat Minus Communications

This configuration is an excursion to the baseline A Sat. It examined the impact to removing the communication hardware associated with the broadcast for GRB and auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.2.

Table 8.2 summarizes in a top-level breakdown the mass and power of the communication subsystem.

The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

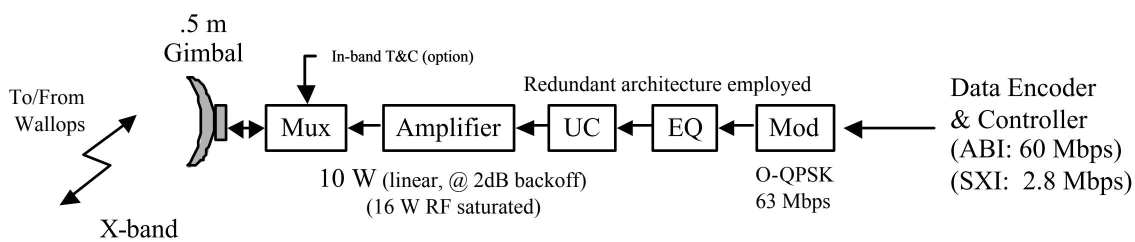

Figure 8.2. Baseline A Sat minus communications system block diagram.

Table 8.2. Baseline A Sat Minus Communications Summary

Baseline A Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink RF Hardware	X-band	16.7	60.0
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22.0
Broadcast System RF Hardware (GRB & Aux)		0.0	0.0
	Communication System Total:	**82.8**	**88.0**

8.2.3 Configuration 3: Baseline B Sat

The sensors on the baseline B Sat included a HES-1 sensor producing 65 Mb/s of data, a HES-2 sensor producing 2.6 Mb/s of data, and a SEM sensor producing 0.56 kb/s of data. To support this sensor suite, the communication system included a 68-Mb/s sensor data downlink transmission, a 10 Mbps GRB transponded transmission, and an auxiliary signal transponded transmission. This configuration involved the following communication system design characteristics:

- Sensor Data Downlink. X-band downlink at 68 Mb/s. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 10.0 W rf linear (16 W rf saturated) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The uplink signal is received through the same 0.5-m gimbaled antenna used for the sensor data downlink to Wallops. The link analysis showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 14 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of backoff from saturation (36 W rf at saturation) was required, when using O-QPSK modulation, for operation of the TWTA. The design of this broadcast system is the same as the baseline A Sat design.

- Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES-R satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W RF (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point. The design of this broadcast system is the same as the baseline A Sat design and, as mentioned, includes the mass and power associated with the auxiliary signal SSPAs.

A conceptual overview of this communication configuration is provided in Figure 8.3.

Table 8.3 summarizes in a top-level breakdown the mass and power of the communication subsystem.

As mentioned, either the baseline A Sat or baseline B Sat spacecraft would have an active GRB and auxiliary broadcast transmission. The payload communication subsystem includes the associated DC power for this capability to ensure that both spacecraft buses are sized to handle the worst-case mass and power load. The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

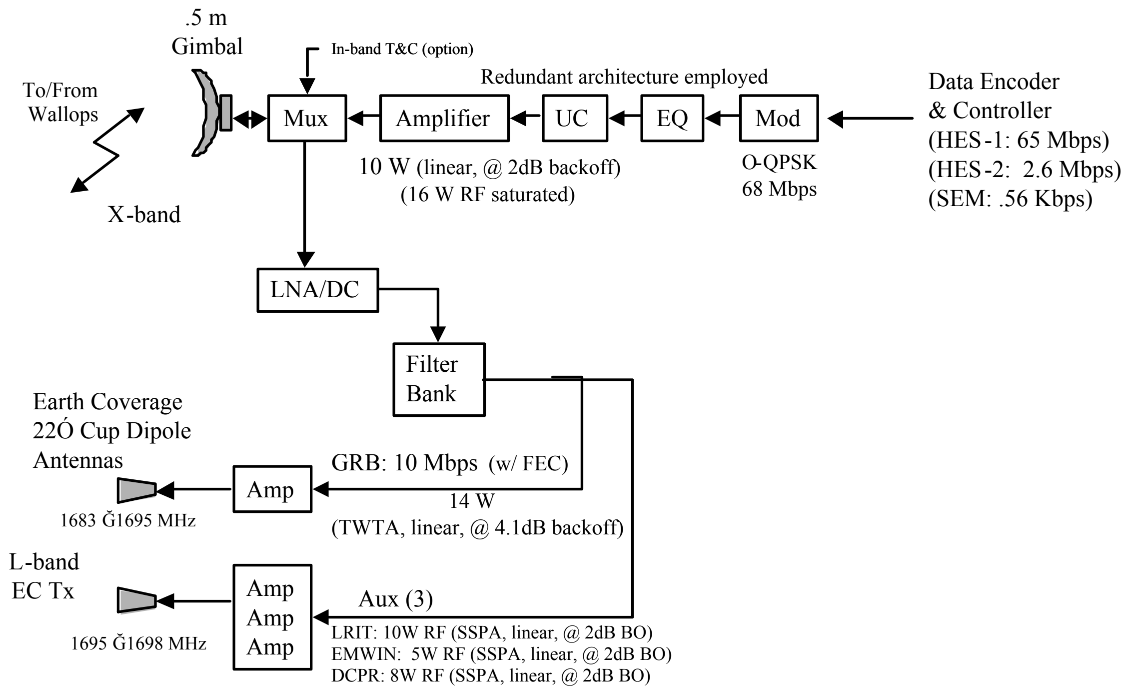

Figure 8.3. Baseline B Sat communications system block diagram.

Table 8.3. Baseline B Sat Communication Summary

Baseline B Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink RF Hardware	X-band	16.7	60
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22
Broadcast System RF Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	208
	Communication System Total:	**155.0**	**296**

8.2.4 Configuration 4: Baseline B Sat Minus Communications

This spacecraft is an excursion to the baseline B Sat spacecraft. It examined the impact to removing the communication hardware associated with the broadcast for GRB and auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.4.

Table 8.4 summarizes in a top-level breakdown the mass and power of the communication subsystem.

Since the baseline A Sat and B Sat sensor data downlink communications systems are the same, the excursions to these systems resulted in the same communication system. The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

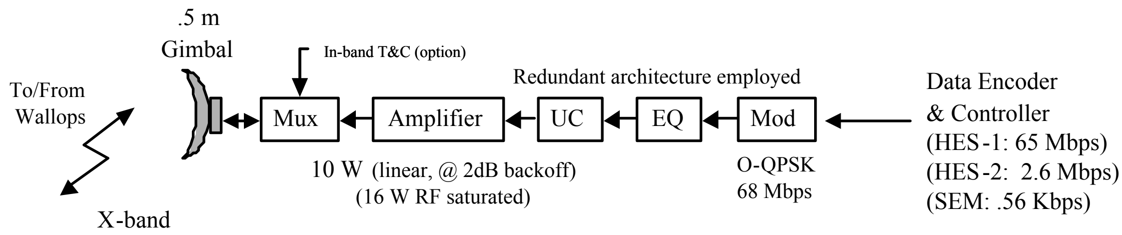

Figure 8.4. Baseline B Sat minus communications system block diagram.

Table 8.4. Baseline B Sat Minus Communications Summary

Baseline B Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink RF Hardware	X-band	16.7	60
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22
Broadcast System RF Hardware (GRB & Aux)		0.0	0.0
Communication System Total:		**83**	**88**

8.2.5 Configuration 5: B Sat MIT

This configuration involved an MIT payload suite whose combined sensor data downlink rate is identical to that of the baseline B Sat payload suite. Therefore, the communication systems are identical. As in the baseline B Sat, the communication system design characteristics include:

- Sensor Data Downlink. X-band downlink at 68 Mb/s. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 10.0 W rf linear (16 W rf saturated) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The uplink signal is received through the same 0.5-m gimbaled antenna used for the sensor data downlink to Wallops. The link analysis showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 14 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of backoff from saturation was required, when using O-QPSK modulation, for operation of the TWTA. The design of this broadcast system is the same as the baseline A Sat design.

- Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES-R satellite to the user community. Link analysis showed that the LRIT signal required 10 W RF (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point. The design of this broadcast system is the same as the baseline A Sat design and, as mentioned, includes the mass and power associated with the auxiliary signal SSPAs.

A conceptual overview of this communication configuration is provided in Figure 8.5.

Table 8.5 summarizes in a top-level breakdown the mass and power of the communication subsystem.

Since the sensor data downlink rate is equivalent to the baseline B Sat, the communication systems are identical. The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

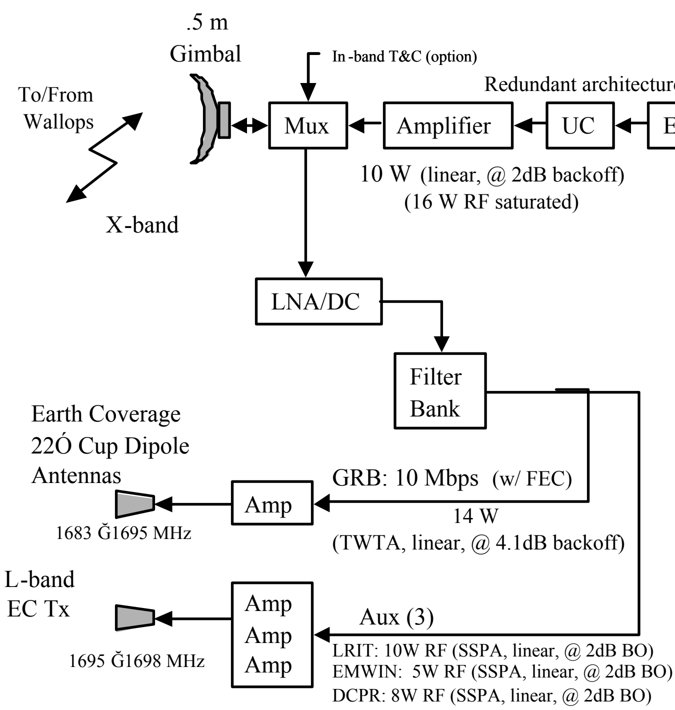

Figure 8.5. B Sat MIT communication system block diagram.

Table 8.5. B Sat MIT Communication Summary

MIT B Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink RF Hardware	X-band	16.7	60
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	34.7	22
Broadcast System RF Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	208
	Communication System Total:	**155.0**	**296**

8.2.6 Configuration 6: Baseline C Sat

The sensors on the baseline C Sat included a GEO STAR sensor producing 2 Mb/s of data, and a Lightening Mapper sensor producing 0.2 Mb/s of data. To support this sensor suite, the communication system included a 2.2 Mb/s sensor data downlink transmission, a 10 Mb/s GRB transponded transmission, and an auxiliary signal transponded transmission. This configuration involved the following communication system design characteristics:

- Sensor Data Downlink. X-band downlink at 2.2 Mb/s. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 0.5 W RF linear (1 W rf saturated) SSPA and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The uplink signal is received through the same 0.5-m gimbaled antenna used for the sensor data downlink to Wallops. The link analysis showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 14 W rf (linear) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of backoff from saturation was required, when using O-QPSK modulation, for operation of the TWTA. The design of this broadcast system is the same as the baseline A Sat design.

- Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES-R satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point. The design of this broadcast system is the same as the baseline A Sat design and, as mentioned, includes the mass and power associated with the auxiliary signal SSPAs.

A conceptual overview of this communication configuration is provided in Figure 8.6.

Table 8.6 summarizes in a top-level breakdown the mass and power of the communication subsystem.

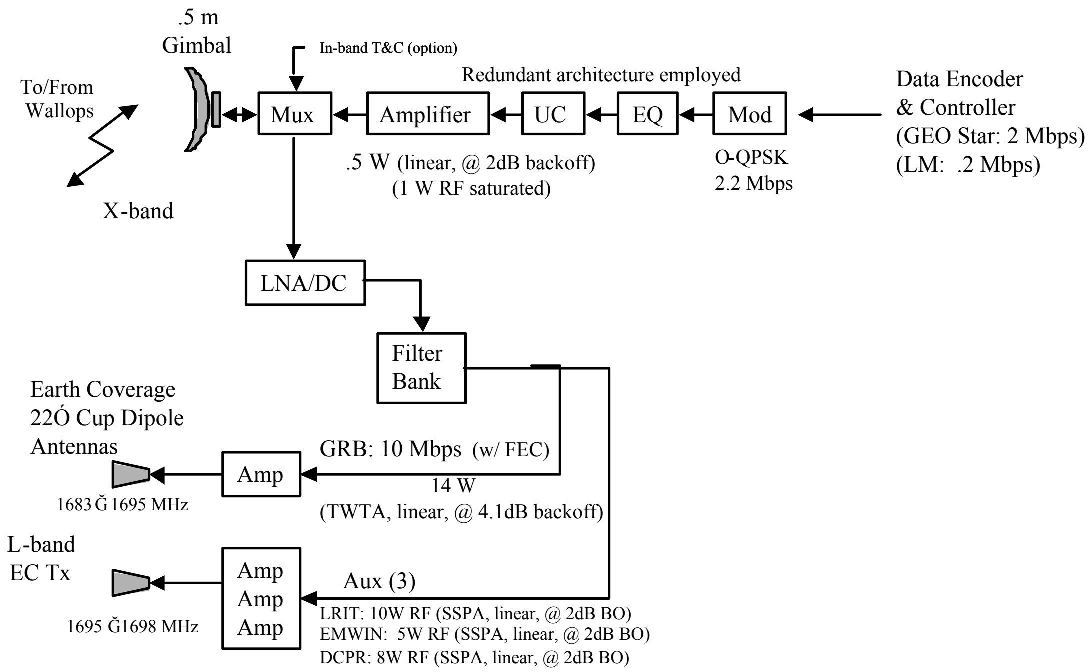

Figure 8.6. Baseline C Sat communication system block diagram.

Table 8.6. Baseline C Sat Communication Summary

Baseline C Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5m GDA	31.4	6.0
Broadcast Antenna Systems	Cup Dipoles	17.6	0.0
Sensor Data Downlink RF Hardware	X-band	7.9	9.0
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	18.6	15.0
Broadcast System RF Hardware (GRB & Aux)		54.6	208
	Communication System Total:	**130**	**238**

The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

8.2.7 Configuration 7: Baseline C Sat Minus Communications

This spacecraft is an excursion to the baseline C Sat. It examined the impact to removing the communication hardware associated with the broadcast for GRB and auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.7.

Since the sensor data rate is only 2.2 Mb/s and GRB service is not required, the 0.5-m gimbaled reflector was replaced by an Earth coverage horn. The reduction in antenna-gain was compensated through the use of a 7-W linear (11 W rf saturated) solid-state power amplifier.

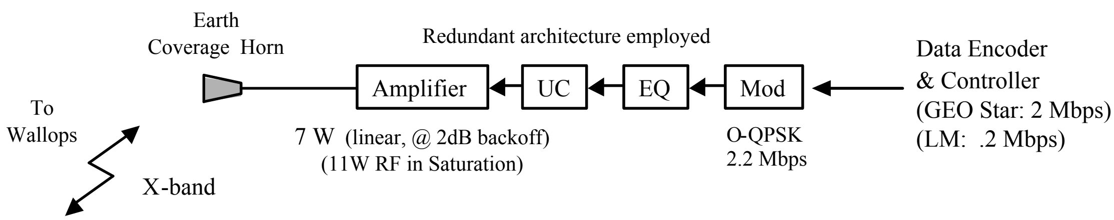

Figure 8.7. Baseline C Sat minus communications system block diagram.

Table 8.7 summarizes in a top-level breakdown the mass and power of the communication subsystem.

Table 8.7. Baseline C Sat Minus Communications Summary

Baseline C Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	EC horn	7.7	0.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink RF Hardware	X-band	16.1	43
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	18.6	15
Broadcast System RF Hardware (GRB & Aux)		0.0	0.0
Communication System Total:		**42.4**	**58**

The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

8.2.8 Configuration 8: Baseline AB Sat

The sensors on the baseline AB Sat included a HES-1 sensor producing 65 Mb/s of data, a HES-2 sensor producing 2.6 Mb/s of data, a SEM sensor producing 0.56 kb/s of data, a ABI sensor producing 60 Mb/s of data, and a SXI sensor producing 2.8 Mb/s of data. To support this sensor suite, the communication system included a 130-Mb/s sensor data downlink transmission, a 10-Mb/s GRB transponded transmission, and an auxiliary signal transponded transmission. This configuration involved the following communication system design characteristics:

- Sensor Data Downlink. X-band downlink at 130 Mb/s. This direct downlink to Wallops was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 20 W rf linear (31 W rf saturated) and a 0.5-m gimbaled antenna to provide 4 dB of link margin to the Wallops receive system. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation. The link margin was increased to 4 dB from the baseline of 3 dB to account for anticipated losses in the X-band diplexer implemented within the space-ground communication system.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The uplink signal is received through the same 0.5-m gimbaled antenna used for the sensor data downlink to Wallops.

The link analysis showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 14 W rf (in linear region) power through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. Previous analysis had indicated that 4.1 dB of backoff from saturation was required, when using O-QPSK modulation, for operation of the TWTA.

- Auxiliary. X-band/L-band uplink/broadcast. As described, this signal set is transponded by the GOES-R satellite to the user community. Link analysis showed that the LRIT signal required 10 W rf (linear), the EMWIN signal required 5 W rf (linear), and the DCPR signal required 8 W rf (linear) through a planar cup dipole antenna providing 15.3 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point. As in the previous configurations, the communication system accounts for the mass and power of the SSPAs for the individual three auxiliary signal transmissions.

A conceptual overview of this communication configuration is provided in Figure 8.8.

Table 8.8 summarizes in a top-level breakdown the mass and power of the communication subsystem.

The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

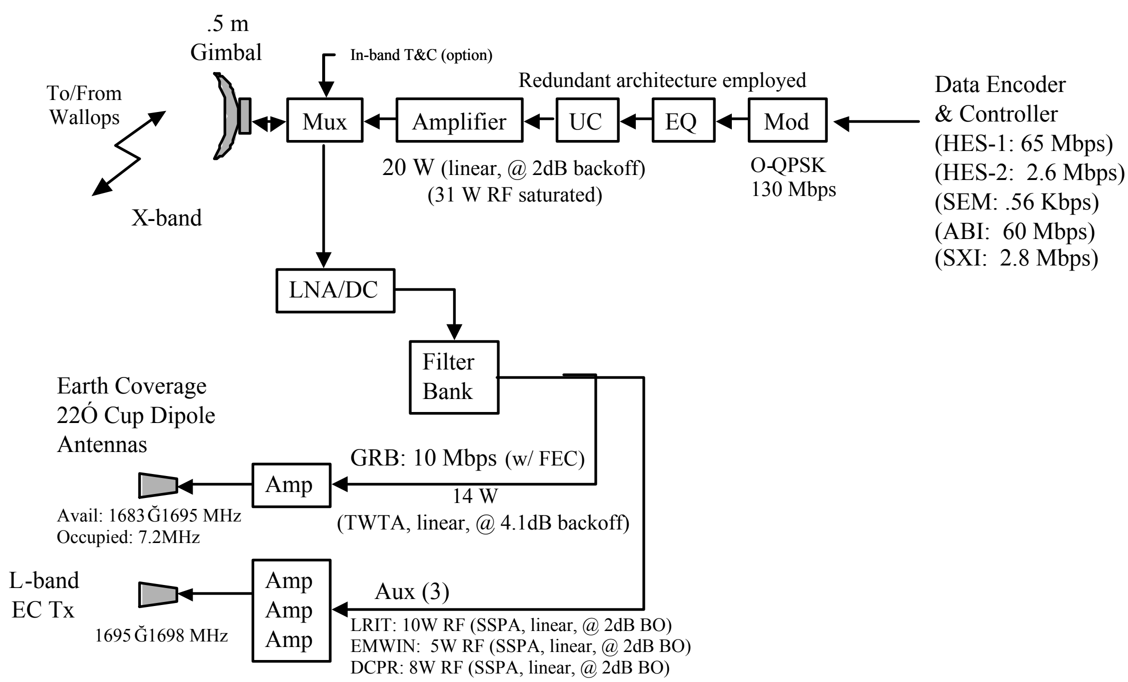

Figure 8.8. Baseline AB Sat communication system block diagram.

Table 8.8. Baseline AB Sat Communication Summary

Combined A+B Sat	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems	Planar Cup Dipoles	17.6	0.0
Sensor Data Downlink RF Hardware	X-band	22	109
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	48.5	30
Broadcast System RF Hardware (GRB & Aux)	L-band O-QPSK (GRB)	54.6	208
	Communication System Total:	**174**	**353**

8.2.9 Configuration 9: Baseline AB Sat Minus Communications

This spacecraft is an excursion to the baseline AB Sat. It examined the impact of removing the communication hardware associated with the broadcast for GRB and auxiliary signals. A conceptual overview of this communication configuration is provided in Figure 8.9.

Table 8.9 summarizes in a top-level breakdown the mass and power of the communication subsystem.

The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

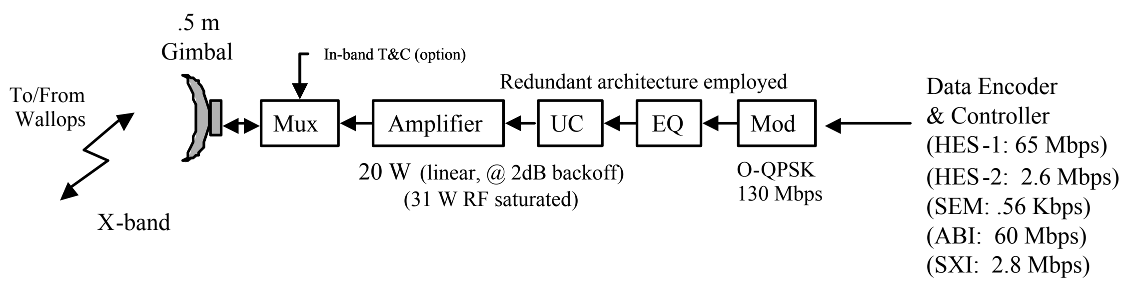

Figure 8.9. Baseline AB Sat minus communications system block diagram.

Table 8.9. Baseline AB Sat Minus Communications Summary

Combined A+B Sat Excursion	Note	Mass (lb)	Power (W)
Wallops Antenna System	0.5 m	31.4	6.0
Broadcast Antenna Systems		0.0	0.0
Sensor Data Downlink RF Hardware	X-band	22	109
Sensor Data Downlink Electronics Hardware	Single pol O-QPSK	48.5	30
Broadcast System RF Hardware (GRB & Aux)		0.0	0.0
	Communication System Total:	**102**	**145**

8.2.10 Configuration 10: Baseline MEO Sat

The concept of the communication system on the baseline MEO Sat is to provide global, as opposed to hemispherical, broadcast of the processed sensor data and the auxiliary signals. The sensor on the MEO Satellite is a GEOSTAR sensor producing 2 Mb/s of data. To support this sensor suite, the communication system included two 2-Mb/s sensor data downlink transmission, a 10-Mb/s GRB transponded transmission, and an auxiliary signal transponded transmission. The communication system includes the following characteristics:

- Sensor Data Downlink. X-band downlink at 2 Mb/s. This direct downlink to defined ground stations (Wallops, and TBD other locations) was designed as a single polarization transmission using O-QPSK modulation with 15/16 FEC. Detailed link analysis defined a transmit power of 1.8 W rf linear (3 W rf saturated) SSPA and a gimbaled horn antenna to provide 4 dB of link margin to the receive system. The linear operation of the X-band SSPA was defined at 2 dB backoff from saturation. Due to the delta in path loss associated with the 11,000 km MEO orbit vs. the 35,000 km GEO orbit, the use of a gimbaled horn antenna system was feasible.

- GRB. X-band/L-band uplink/broadcast at 10 Mb/s. The uplink signal (multiplexed GRB and Auxiliary signals transmitted from Wallops) is received through the gimbaled horn antenna system. The link analysis for the MEO orbit showed that operation in the available downlink broadcast spectrum of 1,683 to 1,695 MHz required O-QPSK modulation, 15/16 FEC, and 10.4 W rf (in linear region, 27 W rf saturated) power through a planar cup dipole antenna providing 8.9 dBi of gain at 5° elevation. The broadcast antennas are smaller in size (9 in. dia) as compared to the GEO application (22 in. dia) in order to provide the same ground coverage as obtained from GEO. As such, their associated gain is less that the GEO versions by approximately 6.8 dB. But this delta in performance is offset by the delta in path loss of a GEO spacecraft versus a MEO spacecraft (8 dB). Previous analysis had indicated that 4.1 dB of backoff from saturation was required, when using O-QPSK modulation, for operation of the TWTA.

- Auxiliary. X-band/L-band uplink/broadcast. As described, the LRIT/EMWIN/DCPR signal set is transponded by the GOES-R satellite to the user community. The link analysis for the MEO orbit showed that the LRIT signal required 7 W rf (linear), the EMWIN signal required 4 W rf (linear), and the DCPR signal required 6 W rf (linear) through a planar cup dipole antenna providing 8.9 dBi of gain at 5° elevation. In all cases, linear operation of the SSPAs was at a 2 dB backoff point.

- Auxiliary One and Auxiliary Two. X-band/L-band uplink/broadcast. The MEO communication system was also required to support the transmission of two additional auxiliary signals in broadcast. The transmission requirement, as defined by the customer, for these signals was defined as near-equivalent rf power associated with the transmission of the LRIT/EMWIN/DCPR signal set. That signal set required 17 W linear rf power (at 2 dB backoff point), therefore, the auxiliary one and two signals were defined to operate at 15 W linear rf power. The mass and power allocated in the communication system is for the

SSPA required for these transmission levels. The payload suite contains the allocation for the transceiver electronics mass and power.

A conceptual overview of this communication configuration is provided in Figure 8.10.

Table 8.10 summarizes in a top-level breakdown the mass and power of the communication subsystem.

Due to the radiation environment at the anticipated MEO orbit, additional shielding was applied to the electronics hardware supporting the sensor data system and GRB system. The on-orbit mission life is supported by the implementation of a fully redundant rf and electronics hardware design.

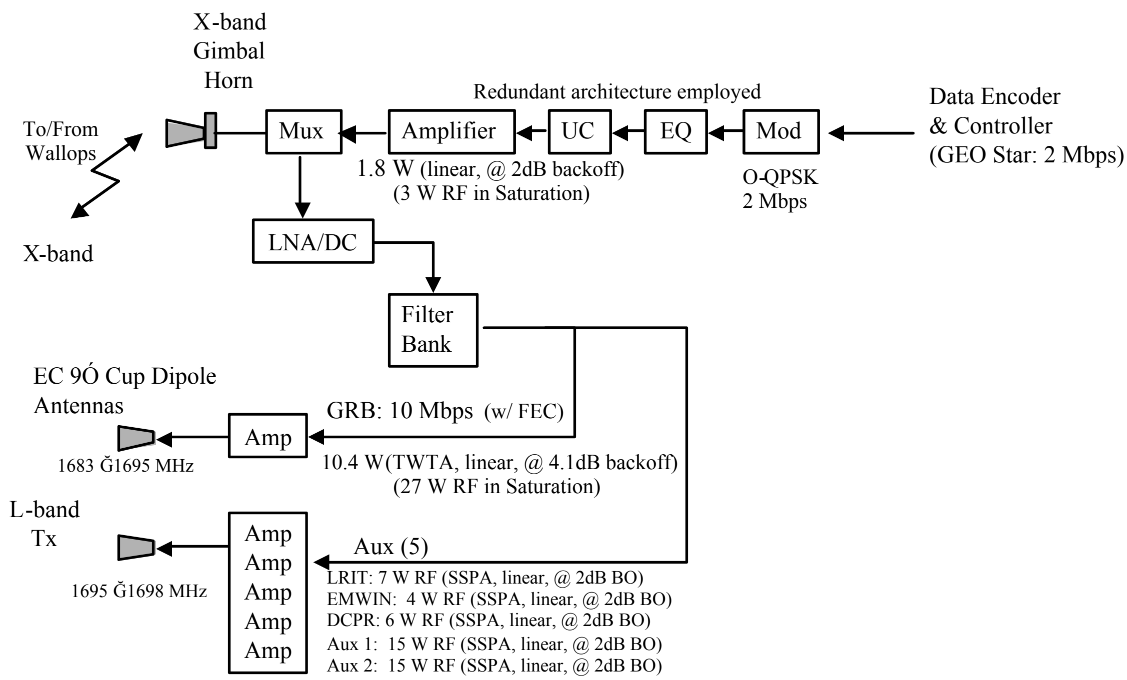

Figure 8.10. Baseline MEO Sat communication system block diagram.

Table 8.10. Baseline MEO Sat Communication Summary

MEO Sat	Note	Mass (lb)	Power (W)
Ground Antenna System	Gimbal Horn	13	4.0
Broadcast Antenna Systems	Planar Cup Dipoles	8.8	0.0
Sensor Data Downlink RF Hardware	X-band	11.2	15.7
Sensor Data Downlink Electronics Hardware		21.4	14.7
Broadcast System RF Hardware (GRB & Aux)	L-band O-QPSK (GRB)	93.2	341
	Communication System Total:	**147.6**	**375**

8.2.11 Configuration 11: Baseline MEO Sat GEO to MEO Insertion

Same as Configuration 10.

8.2.12 Configuration 12: A Sat D1

This configuration is an excursion to the baseline A Sat. It examined the impact of removing the SXI sensor from the sensor suite. The impact to the communication subsystem was seen in the required data rate of the sensor data downlink (SDD); the data rate associated with this excursion was 60 Mb/s versus the baseline 63 Mb/s. This minor reduction in data rate did not justify a change in the design of the baseline A Sat communication subsystem. Therefore, the results reported in Table 8.1 apply for this excursion. As mentioned in the baseline A Sat design description, the communication subsystem was compatible in design with the baseline B Sat design in order to save in non-recurring costs. The same intent is applied to baseline A excursion two and Baseline B excursion two.

8.2.13 Configuration 13: B Sat D2

This configuration is an excursion to the baseline B Sat. It examined the impact of removing the SEM sensor from the sensor suite. The impact to the communication subsystem was seen in the required data rate of the SDD; the required data of the SEM sensor was only 0.56 kb/s and its removal does not change the baseline design rate of 68 Mb/s. Therefore, the results reported in Table 8.3 apply for this excursion.

8.3 Payload Communications Summary

Based on the configuration under examination, the payload communications subsystem provides either the capability to transmit raw sensor data directly to Wallops Island or another NOAA-designated ground segment for the MEO cases, provides the capability to globally rebroadcast processed data and auxiliary data signals received from Wallops Island to globally distributed users, or the combination of both functions. The three spacecraft in each of the GEO orbital slots (75° W and 137° W) frequency share the X-band spectrum (8,215–84,00 MHz) for their direct downlink transmissions. One of the satellites in the orbital trio provides the active rebroadcast capability.

The design of the GRB system is the same for each spacecraft: globally transpond 10 Mb/s of processed data in the L-band spectrum of 1,683–1,695 MHz; and globally transpond three auxiliary signals in the L-band spectrum of 1,695–1,698 MHz. Using O-QPSK modulation and 15/16 FEC, the 10 Mb/s of processed data occupies 7.2 MHz of the allocated L-band spectrum. The DC power requirements of the amplifiers for each of these links were based on operation in the linear region (2 dB backoff for the mission data link, 4.1 dB backoff for the GRB broadcast, and 2 dB backoff for broadcast of the auxiliary signals). The selection of these spectrum bands and these amplifier backoff levels are based on recent study activities currently being conducted within the Communications Systems Subdivision at The Aerospace Corporation in support of NOAA.

9. Attitude Determination And Control

Andrei Doran

9.1 Overview

The GOES-R payloads, especially the ABI, require tight attitude determination (AD). This led to 3-axis stabilized bus attitude control architecture. It is the only approach capable of providing a stable enough platform to permit the required AD accuracy. The attitude determination and control subsystem (ADACS) design is fairly similar across all of the spacecraft configurations, with the only difference being a small mass change in the reaction wheels (RW). The RW sizing was done more accurately in this study because the customer wanted to ensure that momentum unloads are spaced more than two days apart.

AD accuracy was the most important issue among several ADACS areas requiring attention for this mission. The requirement, 3 arc-sec, was stringent, but does not push the AD state of the art envelope set by satellites such as Chandra (where Ball star sensors achieved sub arc-second 3σ accuracy). Discussions with the customer during the study indicated that 7 arc-sec would also be acceptable for the AD requirement. However, since it did not lead to a different sensor selection, the 3 arc-sec requirement was used to satisfy the ABI specifications.* The only difference between a 3 arc-sec sensor and a 7 arc-sec one is the amount of testing done to the unit on the factory bench, which translates to a cost difference. The mass and power are the same in both cases. This is discussed in more detail in the baseline B Sat section, where there is no ABI and the HES driven specification is 7 arc-sec.

An analysis of the combined pointing knowledge obtainable with the best star sensors and gyros showed that the 3 arc-sec specifications can be met. The analysis used the same tool introduced in the first GOES-R CDC study.† This tool calculated the combined error variance based on the individual accuracy of all the sensors. However, the tool has been refined subsequently, and it provided more accurate values for this study. The sensor selection and analysis are described in the AD section of this report.

A related issue was the jitter control and knowledge. There was the usual notation problem, in that jitter specification was not stated as a function of frequency, but rather as a total specification on angular rate. The ABI drives both the jitter control specification and the knowledge requirement. The jitter control value was well within the sensing capability of the selected gyros, and achievable with normal control algorithms and actuator dynamics. However, the knowledge requirement will be tough to meet, unless the already small gyro bias can be calibrated and reduced significantly with star sensor data in the Kalman filter. A resolution of this issue requires further study, and will be discussed at the end of the ADACS report, in the Other Design Considerations section.

---

* T. Kenney, P. Mason and E. Stoneking, GOES-R Study 3, Final Version, October 7, 2002.

† National Oceanic and Atmospheric Administration Geostationary Operational Environmental Satellite-R (GOES-R) Concept Design Center Space Segment Team Study, ATR-2002(2331)-1

Another issue was the single-wing solar panel design. While convenient mechanically (fewer gimbals, cables, etc.), it lead to a more imbalanced structure, with a larger center-of-mass (CM) to center-of-pressure (CP) distance, and incurs a larger solar pressure torque than would a two-wing solar panel design. The issue was not overwhelming, and did not drive the RW size since the accumulated momentum from solar pressure torques over an orbit was less than the momentum needed for the slew maneuver. The slew requirement, a 180° yaw flip twice a year for thermal and solar panel sun viewing reasons, generated approximately twice the momentum accumulation than that from solar pressure during an orbit. The cyclic nature of the solar torque for a nadir-pointing satellite in GEO also helps. The RWs had to handle the accumulation of half an orbit since the next half cancels out the first. Only a small part of the solar torque is cumulative, so the amount of fuel needed to unload momentum is also small. The customer mentioned 10% accumulation as a figure of merit, so this value was used in the calculations. In general, slew maneuvers are much more stressing than environmental disturbances, but here they were only a factor of 2 apart. This was because the slew maneuver can be performed slowly, in 30 min, and because the solar pressure torque was high with the one-wing design. The actuator selection, slew needs, and propulsion requirements for momentum dumping are discussed in the attitude control section.

9.2 Design Summary

The pointing knowledge (aka AD) requirement is the same for all configurations because it is derived from the ABI payload needs. The latest ADACS requirements were taken from T. Kenney, P. Mason and E. Stoneking, GOES-R Study 3, Final Version, October 7, 2002. The document guiding the previous design (A. D. Reth, ACS_req_rev_A'8-9-01.doc, Draft ACS Requirements Document, 8/17/01) listed two sets of requirements, a threshold set and a goal set . The goal then was 1.4 arc-sec, and the threshold was the same 3 arc-sec as now. Table 9.1 summarizes the ADACS design.

The design for the baseline A Sat is considered the nominal case and is described first. The ADACS changes for the other five configurations are listed at the end of this section.

Table 9.1. ADACS Requirements

Design Parameter	Requirement
Stabilization Type	3-Axis Control
Attitude Determination (3σ)	3 arc-sec = 0.0008° = 14 μ-rad
Attitude Control (3σ)	30 arc-sec = 0.008° = 145 μ-rad
Slew Requirements	180°/30 min twice per year
Jitter Requirements	Listed as jitter control and knowledge

9.3 Attitude Determination

The backbone attitude sensors are the star trackers. Three arc-sec 3σ is within the capability of the best star sensors on low-Earth satellites that rotate faster (assuming nadir pointing, thus one rotation per orbit) and get more stars through the star tracker FOV. The nadir pointing geostationary GOES spacecraft stay longer on the same stars, having, in effect, fewer independent star measurements. They dwell longer on the same pixels of the charged coupled device (CCD) sensing plane, allowing a slower smoothing of the pixels centroid position. On the positive side, the high geosynchronous altitude reduces the false star hit errors from the South Atlantic anomaly, but this was a smaller effect.

The slow rotational rate was a larger effect, but it is still possible to meet the 3 arc-sec specification with highly tested star sensors.

A side-analysis during the 2001 GOES-R study showed that it was even possible to meet the stringent 1.4 arc-sec goal most of the time, and perhaps even all the time. That study was done to find the AD accuracy with three highly calibrated Ball CT-602 star sensors and a SIRU. The study used a new tool added at that time to the CDC ADACS spreadsheet. This tool is based on a method developed by Rick Dolphus (R. M. Dolphus, "Simplified Kalman Filters for Control Analysis," Aerospace Technical Memorandum, ATM-99(9990)-3, July 29, 1999) to calculate the combined effect of star trackers and gyros, and also includes the effect of other position sensors. The tool has been refined in many ways, subsequently, the main one being the distinction between the star sensor noise and bias error components. The Kalman filter uses the gyro to take out the star sensor noise, but the bias is unobservable to the filter. Using the upgraded tool, it was shown that 3 arc-sec 3σ is still achievable, including the star sensor bias.

The 2008 technology freeze date permits an even more confident statement than could be made based on the 2004 date for the 2001 GOES-R CDC study since Ball claims it is testing star sensor prototypes twice as accurate as the currently available models used in the side analysis with the combined sensors tool.

A set of three Ball CT-602 star sensors was selected. They are the best Ball models, and with the usual factory testing and calibration have a 3 arc-sec 3σ sigma accuracy specification. Discussions with Ball indicated they can produce such sensors now, and are working on sub arc-sec prototypes. The CT-602 mass and power were used in the instrument list. The three sensors would be placed on the ABI instrument, arranged as the sides of a tetrahedron. They would point South of the orbit plane at an angle larger than 23° (so as to avoid sun impingement). The angle should be no larger than necessary to avoid the sun, to provide the largest possible effective FOV rotation rate, and capture more stars. This arrangement provides 3-axis accuracy with a small reduction in case of failure of one of the sensors. An optimal geometric arrangement of the star sensor directions to provide the highest accuracy in the ABI pitch and roll directions by taking into account the effective rotation rate and star availability was beyond the scope of the study. Placing the star sensors on the ABI payload minimizes the misalignment effects. A 3 arc-sec budget, with nearly that value already consumed by the sensor errors and Kalman filter performance, allows very little room for misalignments since the two sources are root sum squared (RSS) in the total result. It is hard, or impossible, to keep the on-orbit misalignments (thermal, jitter induced, etc.) between the star trackers and the ABI reference frame small enough (approximately less than 1 arc-sec) unless the star trackers are on the ABI, or very near the ABI on a rigid platform.

Other star sensors were mentioned during the 2001 GOES-R CDC study and are worthy of further investigation. The Lockheed AST Mini "1 arc-sec" sensor, basing its high accuracy on more stars in the FOV, seems to have had problems and is not currently supported. Valley Forge Composite Technologies (VFCT) claims it can produce 1 arc-sec star trackers at reasonable prices (approximately three million dollars for four units). However, the current VFCT star sensors on the International Space Station have lower accuracy, so the high-accuracy models have not been proven yet. The selected star sensors, Ball CT-602, and all the other ADACS instruments are described in Table 9.2.

Table 9.2. ADACS Equipment List

Instrument	Units	Unit Mass (kg)	Unit Power (W)	TRL	Comments
Fine Sun Sensors	2	0.05	0.1	8	ADCOLE, 0.017(, 2-axis analog
Coarse Sun Sensors	3	0.08	0.1	8	ADCOLE 2°, 2-axis analog, 180° FOV
Star Sensors	3	5.4	10.0	6	3 arc-sec 3(Ball CT-602, tested
Gyros	2	4.5	20.0	7	Litton SIRU, 4 HRG, low bias
GPS Receivers	2	0.2	4	6	Rockwell Collins NavStrike, light
Reaction Wheels	4	8.5	17	8	Honeywell Constellation HR12, 0.2 N-m, 20 N-m-s
Thrusters	8	See note	See note	N/A	0.5-m moment arm
Nutation Dampers	3	1	0	9	UCN Aerospace, jitter control
ADACS Computer	0	0	0	N/A	Shared with C&DH
Interface Electronics	N/A	2	10	9	Cables, connectors, boards
Total ADACS	17	65	138	6	Power based on average use

Note: Thruster mass and power consumption are included in the propulsion subsystem.

The gyros selected were Litton Space Inertial Reference Units (SIRU), using four Hemispherical Resonating Gyros (HRG) for each SIRU. A fair amount of space experience has been accumulated with the SIRU in the last seven years, and there have been no reported failures on orbit. This model was selected because of its low noise and drift rate. A single SIRU might have been selected, without a spare, since it is an expensive (approximately $800K) and robust unit. Having four HRGs provides robustness to single-point failures in many ways. Any set of three HRGs can measure any the angular rates in any three axes. However, to ensure a 10-year mission life and based on discussions about redundancy with the customer, two units were chosen for the design. A single unit selection remains a possible choice.

Alternate units that were considered are the Litton LN-100 IMU based on mechanical rate gyros, or the Litton LN 200 fiber-optic gyros. They have been in use since the early nineties and like the SIRU have low drift and noise. The LN-100 weighs about twice as much as the SIRU, but costs much less (about $100K versus $800K), while the LN-200 has the advantage of being very light. The fairly bulky (approximately 5 kg) Singer Kearfott mechanical SKIRU is another option. Three 2-axis units would make a fully redundant system. The SKIRU has very low bias and drift, is fairly expensive (approximately $800K to $1M), and has a pretty good track record. It did have on-orbit failures, but mainly in its power supply. Those were identified and corrected, and no further failures have been reported.

Sun sensors are used for safe modes and initial acquisition. Two Adcole 12202 fine sun sensors (0.017°) and three Adcole 18394 coarse sun sensors (2°, 180° FOV) were selected. Both models are very light 2-axis analog sensors. The fairly high-accuracy 12202 sensors may be useful for star tracker initialization, but the main reason for their selection was the low mass and reputable source. The choice of using only coarse sun sensors for safe modes, as the star trackers can self acquire (though it takes longer), is also viable.

Two GPS receivers were included for orbit determination. This choice was based on discussions with the customer and considering the 2012 launch date. By that date, the GPS rf beams will likely be wider, so the number of GPS spacecraft seen from GEO will be higher, permitting GPS-based navigation over larger parts of GEO. Even if GPS is only available part of the time, it permits efficient orbit determination and propagation. At present, few or no GEO satellites use GPS for navigation, partly because the orbital position can be detected accurately enough triangulating with beacons from ground tracking stations, and mainly because of the limited current GPS visibility at GEO. The very light Sandia/Rockwell-Collins NavStrike GPS receiver board was selected (two units). It has been developed for launch and space applications, and Rockwell-Collins is a major producer of military receivers.

9.3.1 Attitude Control

The standard way to control the satellite attitude to the required accuracy (30 arc-sec = 150 μ-rad = 0.008°) is with RWs. The standard set of four RWs arranged in a pyramid configuration was selected. A minimum of three RWs are needed to control momentum in three axes, but four RWs are appropriate for the 10-year mission life, in case one fails. RW failures are rare, but not unheard of. In addition to the electronic components and power supplies, failures can also occur in the mechanical bearings and their lubrication systems.

Unloading the accumulated RW momentum is done with thrusters. An all-thruster system with no RWs would be an undesirable design choice. Even if the thrusters were small enough to be able to control to the required accuracy, the amount of fuel needed for a 10-year life would be prohibitive. The number of firing cycles would also exceed thrusters' limits. The RWs control the large daily torque cycle with essentially no fuel penalty. Only the small cumulative component needs to be unloaded with thrusters. Magnetic rods are generally not used for this type of GEO satellite with a single solar panel and a large daily torque cycle. Calculations confirmed that rods would weigh a little more than the fuel needed for momentum unloading during the entire 10-year life, and add hardware and complexity.

The customer asked for RWs with increased momentum storage ability to facilitate operations and permit less frequent momentum unloading procedures (less than once every two or three days). This led to a more accurate RW sizing than in the previous CDC GOES-R studies. Part of that accuracy improvement was a more exact calculation of the moment arm between the spacecraft CM and the CP. Based on configuration drawings and reasonable assumptions, the single solar wing bus CP to CM distance is 2.5 m. Using this value, the factors sizing the RWs are:

Half orbit accumulation:	RWs must have this momentum capacity to keep the spacecraft pointing within the allowed box without help from thrusters
2.5 days accumulation:	RWs must have this momentum capacity to keep the thruster unload operations more than 2.5 days apart
Slew maneuver requirement:	Usually drives RW sizing

It is interesting that the half orbit accumulation exceeds the 2.5-orbit accumulation because the solar pressure is mainly cyclic (see listing below). The momentum build-up during half an orbit is undone by an opposite buildup during the next half orbit. With customer concurrence, we assumed a conservative 10% cumulative 90% cyclic solar torque ratio. For the baseline A Sat, the three requirements are:

Half orbit accumulation:	10.6 N-m-s
2.5 days accumulation:	5.3 N-m-s
Slew maneuver requirement:	16.2 N-m-s

The semi-annual yaw flip is the main driver for the RW size. An angular momentum of 16.2 N-m-s is needed to rotate the spacecraft 180° in 30 min. The maximum solar pressure torque is approximately 0.00025 N-m. Both the torque and the momentum accumulation come mainly from solar pressure, with gravity gradient and magnetic disturbances orders of magnitude below. Had the yaw slew maneuver needs been significantly higher than the daily attitude needs (say, an order of magnitude higher), then one might consider an alternative design with small wheels for environmental disturbances while the slew is carried out with thrusters. Sizing the wheels for 16.2 N-m-s, however, also covers the half-orbit constraint, and gives the orbit operations planners room for flexible unload scheduling.

Honeywell Constellation 8.5-kg wheels, with a 0.2 N-m torque and over 20 N-m-s momentum capacity were selected, providing an adequate safety margin. With this selection, the wheels stay at less than half of their maximum speed during normal orbit operation since they only need to handle the 10.6 N-m-s half orbit momentum build-up. There is even more margin since four wheels cover three directions, so on average there is 30 N-m-s of momentum reserve per axis, unless a wheel has failed. Twice a year, the wheels incur larger momentum when they perform the yaw flip. With all four RWs the 16.2 N-m-s slew maneuver uses half the 30 N-m-s per axis reserve. With one failed wheel, there is a 5 N-m-s margin. The wheel specs are listed in Table 9.2.

Thrusters are used for periodic angular momentum dumping. With a 0.5-m moment arm (i.e., distance between the satellite's CM and the thruster location), a 1-N thruster force level, and two thrusters per axis, 2-s firings per orbit are required. Given the desire for infrequent unloads, a schedule of 5-s firings every 2.5 days might be chosen. Using a safety factor of 2, a total of 15,000 s of accumulated burn time over the mission life is required. Based on this information, the required propellant mass is calculated by the propulsion subsystem. The approximate fuel amount comes out to 64 kg, based on an I_{SP} of 220 lb/s.

Below are some specific comments on the ADACS designs for some of the other spacecraft designs.

9.3.2 Configuration 3: Baseline B Sat

The baseline B Sat ADACS is similar to the baseline A Sat ADACS in most ways, but there are some differences that will be described here.

The B Sat does not have the ABI instrument, and the attitude accuracy driver is HES. The pointing requirement is the same, 30 arc-sec, but the knowledge requirement is less stringent, 7 arc-sec instead of 3 arc-sec. The jitter requirements are also less stringent. There was no change in the sensors suite from the A Sat to the B Sat because it is the same hardware in mass and power that does the job in both cases. The sun sensors and GPS receivers are obviously the same, as they don't pertain to high accuracy AD. The SIRU gyro is the most appropriate unit in both cases. The star trackers, the backbone of the AD system, are the only instruments that might be considered for a change to less accurate units. However, the same Ball CT-602 models are best suited in both cases, 3 arc-sec or 7 arc-sec. In fact, they are the same basic unit for any accuracy below 10 arc-sec.

The difference is in the amount of testing these units need to undergo on the factory bench to achieve the given accuracy. There are significant cost differences for the different levels of testing. For example, the CT 602 for the A Sat may require eight weeks of additional testing (over the regular, off-the-shelf unit production), and may cost \$2.5M per unit. For the B Sat it may require only four weeks of testing, and may cost only \$1.5M per unit. So, there were no changes in the sensor model numbers in the list of Table 9.2, but there was a reduction in quality and cost. This idea was underscored by discussions about the requirements with the customer and the difficulty in choosing between 3 and 7 arc-sec for the AD specifications.

The RW size was kept the same as in the A Sat because the requirements changed too little to trigger a switch.

Another change pertains to the recommended studies post CDC. The jitter requirements are less severe. It is more likely that the calibrated gyro bias will be small enough to meet the jitter knowledge specifications (2 arc-sec over 15 min and 4 arc-sec over 60 min) than in the A Sat case (0.2 arc-sec in 15 min and 0.8 arc-sec in 60 min). However, even in this case further study is recommended, since the non-calibrated gyro bias is not sufficient to meet specifications.

9.3.3 Configuration 5: B Sat MIT

The same set of sensors and actuators of A Sat and B Sat is used in the B Sat MIT configuration. The AD accuracy was assumed to be the same as in the baseline A Sat design, but this assumption should be verified since neither the ABI nor the HES payloads are on the B Sat MIT version. However, the same Ball CT-602 star trackers kept in this configuration, despite relaxed accuracy requirements, would be less tested and less expensive versions of the same model. The mass and power are the same, but the cost is less.

The only change is a slightly smaller wheel set, 7.5 kg per wheel, saving a total of 4 kg. The momentum requirements are nearly the same as for the A and B Sats, but discussions with the customer and a very careful calculation permitted the mass change. A and B Sats were a little more conservative, and the B Sat MIT version is calculated more exactly.

9.3.4 Configuration 6: Baseline C Sat

The sensors and actuators are the same in the C Sat configuration as in the A Sat configuration shown in Table 9.2, with one exception. Due to the smaller inertias, the slew needs can be achieved with smaller reaction wheels. The momentum requirements are:

Half orbit accumulation:	9.0 N-m-s
2.5 days accumulation:	4.5 N-m-s
Slew maneuver requirement:	8.6 N-m-s

In this case the half-orbit requirement exceeds the slew requirements and is the RW size driver. 5.5 kg wheels satisfy the 9 N-m-s requirement with margin. Twelve kg were saved from the A Sat for the four wheels. The savings are not quite linearly proportional to the ratio of momentum requirements, which would be 1.8 (16.2 N-m-s for Sat A versus 9 N-m-s for C Sat). Wheel mass is not linearly proportional to momentum since wheel inertia is the figure of merit, and it depends more on dimensions than on mass.

9.3.5 Configuration 8: Baseline AB Sat

The baseline AB Sat has the same pointing requirements as the baseline A Sat vehicle. Therefore, the sensors are the same. The satellite is much larger, so the yaw flip requires larger RWs. The momentum requirements are:

Half orbit accumulation:	16.9 N-m-s
2.5 days accumulation:	8.4 N-m-s
Slew maneuver requirement:	40.3 N-m-s

Wheels weighing 9.5 kg were selected to accommodate the 40 N-m-s requirement, raising the total ADACS mass by 4 kg over the A Sat. The on-orbit accumulation is handled conservatively by this wheel choice. The slew maneuver is also covered with safety margin, though slightly less conservatively than the A Sat. That is because the Constellation HR12 series is listed between 6 and 9.5 kg per wheel, and the maximum was chosen, rather than going to a different model. The wheels are the only change to the A Sat design.

9.3.6 Configuration 10: Baseline MEO Sat

The lower orbit, reduced accuracy requirements and longer life lead to several ADACS changes in this configuration. Both sensors and actuators were affected.

The MEO Sat pointing requirements were calculated from a 5-km beam over-spray on the ground and the 10,500-km altitude. That led to a pointing requirement of 0.026º (94 arc-sec or 450 μrad), and an assigned 0.01º (36 arc-sec or 175 μrad) AD entry in the pointing budget. Thirty-six arc-sec (compared to 3 arc-sec for A Sat) allows a much less expensive star sensor selection, but is still out of reach for Earth sensors. In a previous GOES study, the customer suggested a 0.03° 3σ AD accuracy requirement. That permitted an Earth sensor-based AD design, with several advantages over star sensors. Cost is still generally less for Earth sensors, though the low-end star trackers are coming down in price and the differences are small. Earth sensors weigh less, and radiation tolerance for the long 12-year life is a major factor in their favor. Further verification of pointing requirements would be needed to determine whether Earth sensors are an option. Given the calculated requirements, three lighter and lower cost star sensors replace the expensive Ball CT-602 models of the A Sat design. The selected units are the 2.4-kg Ball CT-631 models, with minimal bench testing and lowest COTS cost.

The gyros were not changed from A Sat, but the lighter and less expensive Litton LN 200 fiber-optic unit represents a possible alternative class. The increased bias of the LN 200 over the Litton SIRU used on A Sat might be tolerable with the reduced pointing accuracy.

The MEO Sat can use smaller RWs since it has no slew requirements. The reason for the bi-annual yaw flip of the GEO configurations was probably thermal, so the absence of the flip from the MEO design indicates a more robust thermal design. However, there have been cases where it turned out on orbit that the thermal design was insufficient, and yaw maneuvers had to be implemented. It is prudent to have the option of performing a yaw flip in case of surprises, but that option will be left to the thrusters in this design, and the wheels were sized to handle only the orbital momentum accumulation. The calculated requirements are shown below. In this case, the "no more often than two to three days" unload request results in a 10-orbit accumulation, and unlike the 2.5-orbit GEO cases, this is the factor driving the RW size.

Half orbit accumulation:	3.0 N-m-s
2.5 orbits accumulation:	1.5 N-m-s
2.5 days (10 orbits) accumulation:	5.9 N-m-s
Slew maneuver requirement:	0 N-m-s

Honeywell Constellation HR0610 wheels were selected. They weigh 4 kg per unit and meet the 6 N-m-s requirement with margin.

The customer requested a verification of the smaller wheels capacity to handle the motion of an Earth-pointing 1-m-dia parabolic antenna. Even without detailed mass properties data on the antenna, it was possible to rule out this concern. Based on approximate dimensions and masses, the antenna inertia was calculated as 5 kg-m^2. The maximum angular rate needed to track a fixed Earth target is when the target is directly below at nadir, and is 0.0005 rad/s. The angular momentum implied by that inertia and rate is 0.0025 N-m-s, orders of magnitude below the 5.9 N-m-s requirement regardless of approximating assumptions.

The torque needed to provide a stable bus platform during antenna motion is derived from antenna accelerations, and can be a driver in agile applications even if the momentum is negligible. In this case, however, the antenna acceleration is low, less than 6e^{-8} rad/s, requiring 2.8e^{-7} N-m of torque from the RWs. This torque is negligible compared to the 0.2 N-m RW capability, and the chosen wheels can easily handle all the effects of antenna motion.

The customer also requested a check on the desirability of a magnetic rod momentum unloading system. GPS satellites, for example, use magnetic rods to unload momentum at twice the altitude, and the magnetic effect decreases with the cube of the distance from Earth. Comparing magnetic- versus propulsion-based unloading, the important factors are mass and complexity. It turned out the mass was fairly similar. Approximately 18 kg of propellant is needed to unload momentum over the 12-year life, and also about 18 kg of magnetic rods are needed to the cumulative part of the solar pressure. However, magnetic rods add complexity to the system, since the propulsion system is there in any case. In addition, a 0º inclination orbit is not very good for magnetic torque, even though it is relatively close to Earth (compared to GEO or GPS-like half GEO). The orbit is always within 22º of magnetic equatorial, so the magnetic vector variation is very small. Two axes have adequate control authority, but one does not, requiring that two out of the three rods be very large. Therefore, it was decided to unload momentum with thrusters. If there had been no propulsion system in place already (e.g., for station keeping and EOL de-orbit issues), the assessment would probably be the same, but further and more detailed study would be recommended.

In summary, the MEO Sat ADACS is 27 kg lighter than the A Sat ADACS. Eighteen kg savings come from the lighter RWs and 9 kg from lighter star sensors. A few kg further savings might be possible with a lighter gyro selection. If the AD spec can be relaxed from 0.01º to 0.03º, then another few kilograms can be saved by switching from star trackers to Earth sensors.

9.3.7 Configuration 12: A Sat D1

The ADACS is essentially the same as the baseline A Sat. There were minor changes in inertias. That, coupled with a very careful look at the RW sizing, led to a small mass reduction. The same Honeywell Constellation HR12 wheel model were used, but at eight kg per unit as opposed to 8.5 kg in the baseline. This produces a 2-kg total system mass reduction.

9.3.8 Configuration 13: B Sat D2

The ADACS is essentially the same as the baseline B Sat (as well as the baseline A Sat). As in the A Sat case, there were minor changes in inertias, which, coupled with a more careful look at the RW sizing, led to another small mass reduction. Again, the Honeywell Constellation HR12 wheel model was used, but at 7.5 kg per unit as opposed to 8.5 kg in the baseline. This produces a 4 kg total system mass reduction compared to the baseline and a 2 kg reduction compared to A Sat D1.

9.4 Other Design Considerations

Following is a list of assumptions and design issues that are beyond the scope of concept studies. The jitter issues (discussed below) are especially important, and further study in that area is recommended in the subsequent design stages to ensure that the ABI requirements can be met.

- There is no ADACS-dedicated computer. Mass savings are obtained by sharing the C&DH computer, which is in the RAD 6000 class, or better, and has more than enough throughput for the ADACS software, including star catalog calculations.

- Two kg of cables and interfaces are included for Attitude Control Electronics (ACE).

- Jitter requirements are covered in two ways. One part, called "control" requires 20 arc-s/s 3σ. The second part, "knowledge," requires 0.2 arc-s over 15 min and 0.8 arc-s over 60 min, both values 3σ. The ABI is the driver for these requirements. The jitter control requirements are achievable with the precision rate sensor chosen, but the knowledge part requires further study. The SIRU has a 0.008°/h bias, i.e., 0.008 arc-s/s. Calibration in the Kalman filter using the star tracker for reference can reduce the bias. Without those further reductions the knowledge requirement is not met. 0.008 arc-s/s implies 7.2 arc-s/15 min, much more than the 0.2 arc-s specified. Also, it implies 28 arc-s/60 min, much more than the allowed 0.8 arc-s. However, the control part is satisfied. The sensor is within 0.008 arc-s/s of true rate, and the controller needs to stay within 20 arc-s/s. There is enough margin for the errors inherent in controller software and actuator nonlinearities. A simulation is recommended, but the prognosis is optimistic to meet controller specifications. For the jitter knowledge, meeting specifications depends on how much of the bias can be reduced by the calibration process. The answer requires further study, and it seems the state of the art is being pushed.

- A study of the jitter sources is also recommended, given the tight jitter knowledge and control specifications. The solar array drive and the RWs are internal jitter sources, and there may be others associated with thruster firings and active cooling instruments. A jitter study, including a characterization of the jitter sources, and of their structural attenuation between the source and the ABI, is needed before deciding with certainty whether active jitter suppression or filtering is necessary. The assumption made here is that there is sufficient damping and frequency separation between the control loop and the jitter sources that the jitter requirements are met passively. Three nutation dampers were included as a placeholder for passive damping.

This page intentionally blank.

10. Power

Ed Berry

10.1 Overview

The electrical power subsystem (EPS) designs for all GOES-R configurations were based on the following conditions and assumptions:

- Planar, single-wing solar array, one-axis sun tracking, with multi-junction 32% efficient GaAs/Ge solar cells and lightweight Al honeycomb panels.
- Li-ion batteries with energy density of 100 W-h/kg, maximum depth-of-discharge (DOD) of 60% and 50% for 10-year and 15-year GEO missions, respectively; NiH_2 batteries with energy density of 50 W-h/kg and maximum DOD of 50% for MEO Sat.
- Direct-energy-transfer EPS; spacecraft bus is battery regulated and shunt limited to 37.5 ± 5 V, same as 2001 GOES-R study.

10.2 Design Summary

The selection of Li-ion batteries for the GEO configurations is aggressive because they have little flight heritage as yet, although they are under active development and have a fairly large and growing test heritage. Their major attraction is their high energy density, about twice that of NiH_2 batteries. There is now a general perception that Li-ion will ultimately become the standard spacecraft battery type, replacing NiH_2, but the development time necessary to accomplish this is uncertain. It is possible that by 2008, Li-ion technology would not be sufficiently mature to commit to a 10-year mission. In this case, the alternative technology would be NiH_2, and the battery weights for the GEO configurations would be about double those calculated in this study.

For the MEO Sat mission, the battery cycle-life requirements are much more severe than for the GEO missions because of the much greater frequency of eclipses in the lower orbit. The CDC battery cycle-life database indicated that to meet these requirements, the Li-ion battery DOD would have to be reduced so much that NiH_2 batteries would be lighter. Consequently, we assumed NiH_2 batteries with a maximum DOD of 50% for MEO Sat.

The 32% solar cell efficiency was based on projections of current technology and estimates of vendors and others engaged in cell research and development. If cell technology in 2008 cannot provide 32% efficiency, then solar array size and mass would be somewhat higher than calculated. Cell efficiencies currently available are approximately 26 to 28%.

Trapped radiation dosages were based on the JPL GaAs Solar Cell Radiation Handbook, JPL Publication 96-9, updated to reflect recent radiation test results for multi-junction GaAs/Ge cells. Solar proton dosages were based on the JPL 1991 solar proton event model with an 80% probability of not being exceeded. Solar cell cover glass thickness of three mils of fused silica was optimal for all GEO configurations, for minimizing solar array mass. For MEO Sat, the optimal cover glass thickness was 30 mils, reflecting the much more severe radiation environment in the lower orbit; despite the thicker cover glass, the solar array radiation degradation in MEO was substantially greater than in GEO. Solar cell backside shielding was equivalent to about 18 mils of fused silica.

Major EPS parameters for the eight baseline configurations are summarized in Table 10.1. Solar array mass includes allowances for deployment and orientation hardware.

Table 10.1. Power Subsystem Summary

Configuration	A Sat	A Sat D1	B Sat	B Sat D2
Mission Length (yr)	10	15	10	15
Solar Arrays				
BOL Power (W)	2707	2478	2723	2458
EOL Power (W)	2262	1991	2276	1975
Solar Array Area (m^2)	8.1	7.4	8.2	7.4
Solar Array Mass (kg)	44	41	45	40
Batteries				
Type	Li-ion	Li-ion	Li-ion	Li-ion
Maximum Depth of Discharge (%)	60	50	60	50
Total Capacity Req'd (A-h)	98	113	107	112
Battery Mass (kg)	38	43	41	43
Power Mgt. And Dist. (PMAD)				
Wiring Harness Mass (kg)	58	51	58	51
Pwr. Reg. & Cond. Mass (kg)	38	34	38	34
Total Power Subsystem Mass (kg)	178	169	182	168
Mission Length (yr)	10	10	10	12
Solar Arrays				
BOL Power (W)	2645	2282	4661	3379
EOL Power (W)	2210	1907	3894	2099
Solar Array Area (m^2)	7.9	6.8	14.0	10.1
Solar Array Mass (kg)	43	37	76	74
Batteries				
Type	Li-ion	Li-ion	Li-ion	NiH_2
Maximum Depth of Discharge (%)	60	60	60	50
Total Capacity Req'd (A-h)	104	90	175	68
Battery Mass (kg)	40	35	67	54
Power Mgt. And Dist. (PMAD)				
Wiring Harness Mass (kg)	56	50	92	49
Pwr. Reg. & Cond. Mass (kg)	37	33	61	33
Total Power Subsystem Mass (kg)	**177**	**154**	**296**	**209**

11. Propulsion

Keith Coste

11.1 Overview

Two basic types of propulsion systems were investigated: dual-mode hydrazine and bi-propellant using monomethylhydrazine (MMH) with nitrogen tetroxide (NTO). For system configurations that resulted in relatively low total propellant mass requirements, the hydrazine dual-mode system was incorporated. As total mass increased, the system was adjusted to utilize a more conventional MMH/NTO bi-propellant design. During transfer orbit, the specific impulse (Isp) used for this sizing study was assumed to be the same for both system types at a value of 323 s. Although slightly higher values are possible with hydrazine (<2% improvement), an increase in risk, lack of proven heritage, and minimal system benefit drove the decision to utilize the 323-s value for all cases. For the baseline MEO Sat GEO to MEO insertion configuration, an Isp of 325 s was selected for the dual-mode hydrazine transfer orbit. This was an attempt to fully optimize that configuration for minimum mass.

11.2 Design Summary

Other than the direct injection case, transfer orbit propellant usage dominated the propellant budgets for each configuration. There were three factors that directly impacted the transfer orbit propellant consumption calculations. First was the Isp of the orbit raising engine. As described above, this value was held constant for all of the configurations except one. The second factor was transfer orbit delta-V. In this study, the delta-V was assumed to be constant for each orbital destination. The third factor affecting transfer orbit propellant usage was total vehicle mass at the completion of transfer orbit. This had the most variable impact on the overall propellant budget due to vehicle dry mass variations for each configuration and on-orbit station keeping requirements, which resulted in variations in on-orbit propellant usage.

In addition to station keeping requirements, propulsion system Isp was a factor in determining on-orbit propellant usage. This resulted in the sole area of discrimination between the dual-mode hydrazine system and the conventional MMH/NTO bi-propellant system. The dual-mode system operates in a low Isp monopropellant mode during station keeping, while the bi-propellant system can provide approximately 30% improvement in performance. Since 30% less station keeping propellant would be carried to orbit, the transfer orbit propellant usage with the bi-propellant system will be reduced as well. Although this Isp improvement can result in significant overall propellant mass savings, residuals must be considered to find the true benefit of a bi-propellant system over the dual-mode system.

Residuals are defined as the unusable propellant remaining in the spacecraft at EOL and contain two main constituents. First is the inaccessible propellant such as liquid left in the tanks and dead ended lines and propellant vapor throughout the system. Second is the potential for single species (i.e., fuel or oxidizer) remaining after the initial depletion event. Single-species residuals often occur in bi-propellant systems due to variations in system pressures, liquid flow pressure drop, and thruster level mixture ratio biases. Exact estimation of residuals is extremely difficult at this stage, and simplifying assumptions must be made based on prior experience. The results of this analysis indicated that for relatively low on-orbit usage requirements, the residuals in the bi-propellant system outweigh the performance benefit when compared to low-performance but low-residual monopropellant operations of the dual-mode system. Table 11.1 shows the summarized propulsion system mass breakdown, including transfer orbit and on-orbit propellant usage.

Table 11.1. Design Summary for All Configurations

Configuration	System	Propellant Budget: Orbit Insertion (lbm)	Propellant Budget: NSSK (lbm)	Propellant Budget: Total (lbm)	Propulsion Dry Mass (lbm)	Isp for transfer Orbit	Isp for RCS
A Sat minus Comm	Dual Mode	1457	337	1883	137	323	215
A +B Sat minus Comm	Bi-propellant MMH/NTO	2317	424	2909	185	323	280
B Sat minus Comm	Dual Mode	1465	339	1893	137	323	215
B Sat MIT	Dual Mode	1865	432	2409	143	323	215
Baseline A Sat	Dual Mode	1684	390	2169	137	323	215
Baseline B Sat	Dual Mode	1691	392	2179	137	323	215
Baseline C Sat	Bi-propellant MMH/NTO	1228	225	1544	103	323	280
Baseline C Sat minus Comm	Bi-propellant MMH/NTO	1063	194	1335	103	323	280
Baseline MEO Sat	Hydrazine Monopropellant	na	na	45	22	na	215
Baseline Single Sat	Bi-propellant MMH/NTO	2937	537	3688	225	323	280
MEO Sat Transfer	Dual Mode	1347	na	1429	117	325	215
A Sat D1	Bi-propellant MMH/NTO	1705	463	2304	161	323	280
B Sat D2	Bi-propellant MMH/NTO	1625	441	2201	126	323	280

12. Structure

Kenneth Mercer

12.1 Overview

The structures subsystem was sized based upon an empirical approach found to be reasonable in past studies. Structures mass was derived as a specified fraction of spacecraft dry mass. The appropriate mass fraction was established from a combination of historical data and projected capabilities of technologies. The launch vehicle adapter mass was carried at the systems level as a reduction in the vehicle capability. On multiple launch manifests a payload attach fitting (PAF) adapter mass was estimated based upon current Atlas V PAF designs and included in the structures allocation. Mass contingencies were added at the system level rather than the component level.

Estimates of spacecraft bus dimensions were also developed to enable inertia predictions for ADACS sizing. A maximum dimension is determined from the largest square inscribed within the fairing diameter. In most cases, the spacecraft bus width was dictated by the mounting area required by a given configuration's payload suite. Next, an appropriate height factor was determined to aptly contain necessary internal components and to satisfy a specified bus density. A bus density limit of 160 kg/m^3, which does not include payload or solar array mass, was specified. This bound is based upon historical data and is used to address the potential for spacecraft packaging issues. For sizing of the solar arrays, the number of panel segments, each approximately the size of a bus panel, was calculated to meet the total solar array area specified from the power subsystem.

The magnetometer boom was separately sized to meet stiffness requirements. The design lateral frequencies were greater than or equal to 10 Hz for this boom. Balance mass for center-of-gravity control was selected at 1% of spacecraft dry mass.

12.2 Design Summary

The projected structures mass fraction for the current study corresponded to a technology freeze date of 2008. Since use of a commercial bus was preferred, a structures mass fraction of 18% was used. For this type of spacecraft, this mass fraction assumes a large amount of composite materials. The structural mass fraction does not include the additional mass from the payload support structure and boom.

Because of the similarity to commercial-type buses with space heritage, the spacecraft structure is a TRL of seven. The standard LV adapters will be based upon the current adapters; therefore, they are assigned TRLs of six. A custom adapter is required for all multiple launch manifests, resulting in a TRL of five. A launch isolation system offers a potential mass reduction for the dual- and multiple-vehicle missions, but is not included in the current study. Table 12.1 presents a summary of the technology maturity assessments.

Table 12.1. Technology Assumptions

Spacecraft Structure Mass Fraction	0.18
Technology Readiness Levels (NASA TRL)	
Spacecraft Structure	7
Launch Vehicle Adapter	6
Dual or Multiple Launch Adapter	5

The baseline A Sat, B Sat, B Sat MIT, C Sat, and MEO Sat configurations are very similar in their arrangement because they all utilize the top surface of the bus to mount their payloads. The baseline AB Sat contains the full suite of electronics from the A Sat and B Sat; therefore, it requires the addition of a payload module. The payload module is assumed to act as a mini-bus and is sized as a mass fraction of the mounted components. The bus structural mass fraction is reduced to 16% of vehicle dry mass to provide mass credit for the presence of the payload module. Tables 12.2 and 12.3 present the mass summary and structure designs for these configurations.

For all configurations, it was assumed that on-orbit disturbances are mitigated by the ADACS. As a result, no optical bench is utilized to isolate sensitive payload sensors. All configurations exhibit room to fit within the payload fairing. Packaging within the spacecraft is also not a concern since the bus density is well within the allowable limit.

Over the course of the design sessions, all of the baseline configurations were revisited several times with various reductions in payload and/or changes in mission life. Only the baseline AB Sat build without the DCS and SAR provided significant structural impact. In this case, the baseline AB Sat did not require a payload module, as the remaining payloads could be configured directly on the bus.

Table 12.2. Spacecraft Structure Design

	Configurations					
Spacecraft parameters	**A Sat Baseline**	**B Sat Baseline**	**C Sat Baseline**	**AB Sat Baseline**	**MEO Sat Baseline**	**B Sat MIT ver.**
Bus density, kg/m^3	93	94	121	107	92	99
Effective bus dimension, m	2.9	2.9	2.25	3	2.25	2.9
Bus height, m	1.9	1.9	1.9	2.5	1.7	1.9
Payload height, m	1.3	1.3	1.3	4.3	1.3	1.3
Single bus panel area, m^2	5.4	5.4	4.2	7.5	3.8	5.4
No. solar panels per array	2	2	2	3	3	2

Table 12.3. Structure Mass Results

	Configurations					
Mass Breakdown	**A Sat Baseline**	**B Sat Baseline**	**C Sat Baseline**	**AB Sat Baseline**	**MEO Sat Baseline**	**B Sat MIT ver.**
Basic bus structure, kg	218	219	171	359	160	241
Payload module structure, kg	N/A	N/A	N/A	162	N/A	N/A
Mechanism, kg	10	10	10	10	10	10
Balance, kg	15	15	12	22	11	17
Booms, kg	3	3	3	3	3	3
Total structure, kg	**246**	**247**	**196**	**556**	**184**	**271**

In all other cases, the modified baseline satellites had minimal effect on the configuration's structure. Other relevant details of these trade studies are provided in the systems section.

12.3 Recommendations/Issues

Several of the sensors require high pointing accuracy. As a result, there is potential for performance degradation due to structural disturbances. It is not possible to reliably predict structural disturbance at the concept design phase; therefore, this analysis should be performed early in the design phases. One potential for combating dynamic disturbance problems is to incorporate an optical bench, but this will incur a mass penalty. Furthermore, it is expected that the inclusion of an optical bench would have a large impact on current configurations because of the already challenging mounting scheme.

On all configurations, the payload suite includes several sensitive optics and sensors. Contamination control of these components should also be addressed early in the program. The key concern is contamination due to off-gassing of composite materials. This is because bus and payload structures are assumed to be mostly composite material.

It is recommended that all dual-launch manifests utilize dual PAFs. A separate study during the CDC session showed that it was not advantageous to use structure of the inboard vehicle to support an outboard vehicle during launch. In fact, providing a more robust structure on the inboard vehicle revealed a substantial mass penalty.

This page intentionally blank.

13. Thermal

Bill Fischer

13.1 Thermal Overview

The thermal subsystem uses relationships between thermal control parameters and the margined orbit average power and the margined spacecraft dry mass to estimate thermal control mass and heater power. This association is based upon data accumulated from many space programs. The thermal designs of the GOES-R spacecraft employ standard commercial satellite thermal control technology. This includes quartz mirrors on radiators with multi-layer insulation (MLI) blankets on the remaining external structure. Heat pipes and thermal doublers should be used to spread heat out from concentrated heat sources. Heaters will be required for temperature control at beginning-of-life conditions and during cold periods.

Considerable radiator area will be needed to dissipate the energy generated by the payload. Heat pipes will be needed to move the payload heat to the radiators. The spacecraft bus is a cubic structure with each face of the cube approximately 5 to 6 m^2. The spacecraft orbital orientation has one face of the cube nadir facing. Solar panels extend from adjacent cubic panels to the nadir panel and are oriented toward the sun. The nadir-facing panel is the prime payload sensing equipment location. On each of the payload boxes, radiator area may be located to dissipate a portion of the payload heat. Figure 13.1 shows the amount of area required per watt of power dissipation up to 100 W of dissipation.

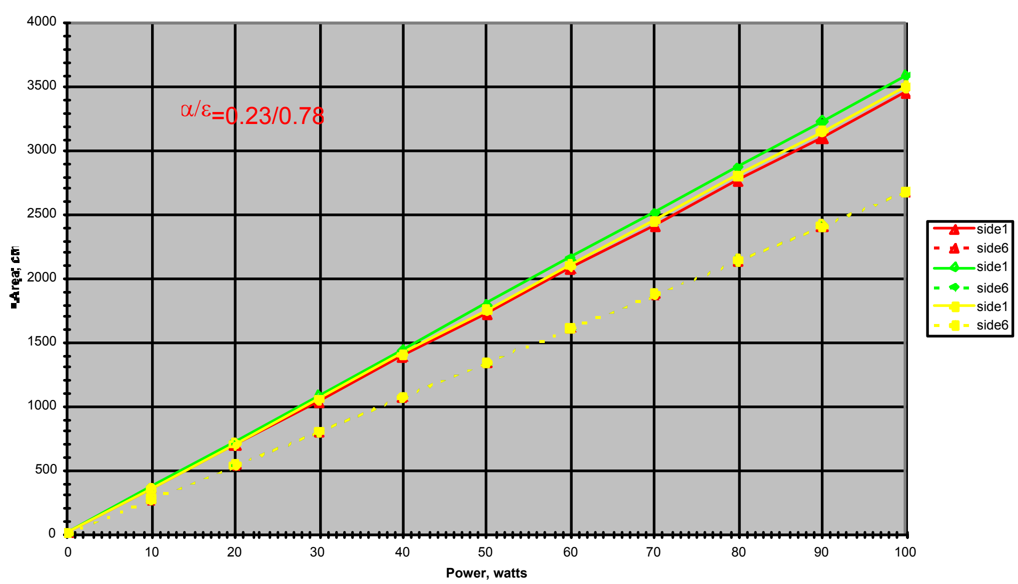

Figure 13.1. Required radiator area for 100 W.

In Figure 13.1, side 1 is the required area for an average face of the vehicle that receives some solar loading during the GEO. Side 6 is the one side of the spacecraft that does not see the sun. For example, this would be the south-facing face of the vehicle when the sun is in the northern hemisphere. Most of the nadir panel is used for equipment location. This leaves approximately five remaining panels to use for radiator locations. Thus, as much as 25 m^2 are available for radiator area. The ABI is slated to use a mechanical refrigerator. This device will use a warm temperature radiator to dissipate the thermal energy generated by the compressor. The original thermal baseline for the ABI included a passive cooler or radiator.

A passive cryogenic radiator is a low-temperature, low-power radiator. The radiator area estimated by Donabedian in *Status and Current Technology of Radiant Coolers** is 1.7 m^2. The passive cooler constrains the vehicle orientation to a 180° flip as the sun passes from the northern to southern hemisphere. This radiator would be located on the side-6 face of the spacecraft. The baseline for the ABI is expected to change to a mechanical refrigerator. A warm-temperature radiator for a mechanical refrigerator radiator has less stringent operational requirements and may not require a seasonal 180° flip. Unfortunately, the refrigerator will require over 100 W of input power to achieve the cooling necessary to cool the ABI.

If the total dissipated power is under 2,000 W, as in all the configurations studied, the required radiator area as shown in Figure 13.2 is 5 to 7 m^2. This radiator area is easily distributed within the available area on the spacecraft body.

13.2 Design Summary

The power in watts required for the thermal control system is shown in Table 13.1.

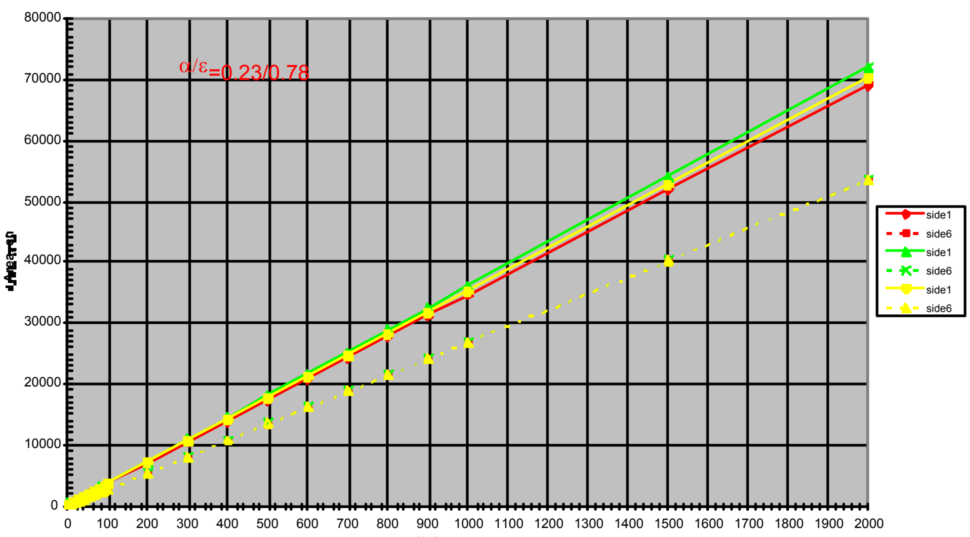

Figure 13.2. Required radiator area for 2,000 W.

* Donabedian, Martin, <u>*Status and Current Technology of Radiant Coolers*</u>, Aerospace Report No. ATR-2001(2331)-2, 24 July 2001

Table 13.1. Thermal Control System Power Requirements, W

Configuration	Thermal	P/L	Prop	ADACS	TT&C	C&DH	Power	Total
A Sat (Baseline)	238	1197	0.2	138	39	15	23	1650
B Sat (Baseline)	238	1196	0.2	138	39	15	26	1652
C Sat (Baseline)	186	984	0.1	138	39	15	22	1384
A+B Sat (Base)	440	2153	0.1	138	39	29	42	2841
MEO Sat Transfer	174	1036	0.9	95	39	16	64	1425
A Sat (Base)-Comm	195	838	0.2	138	39	15	17	1242
B Sat (Base)-Comm	197	838	0.2	138	39	15	19	1246
C Sat (Base)-Comm	158	751	0.1	138	39	15	17	1118
A+B Sat (Base)-Comm	347	1645	0.1	138	39	27	32	2228
B Sat MIT	254	1132	0.2	138	39	15	25	1603
MEO Sat Transfer	198	1005	0.2	95	39	15	54	1406

It is immediately obvious that the power dissipated for propulsion, ADACS, and TT&C are the same for most of the spacecraft designs except the combined AB Sat and the MEO Sat. Only the payload, C&DH, EPS, and thermal subsystem powers are varying. For all of the bus components, simple passive thermal control techniques such as thermal control coatings and heaters are sufficient. Based on the complexity of payload configuration design, constant conductance and/or variable conductance heat pipes may be necessary to spread the high payload power to body-mounted radiators. The primary thermal concern will be the actual thermal design of each payload element and the location of its radiator.

The design parameters for the thermal mass in kg of the configurations for GOES-R are presented in Table 13.2.

The primary difference in these configurations is due to different spacecraft mass values associated with the payload configuration. Maximum payload mass is achieved with the combined A and B satellite payload configuration. The thermal requirements are significantly lower for the remaining payload configurations.

Table 13.2. Thermal Control System Mass, kg

Configuration	P/L	S/C Dry	Thermal
A Sat (Baseline)	520	1208	35
B Sat (Baseline)	520	1214	35
C Sat (Baseline)	373	947	27
A+B Sat (Base)	979	2241	88
MEO Sat Transfer	379	886	26
A Sat (Base)-Comm	437	1041	30
B Sat (Base)-Comm	438	1046	30
C Sat (Base)-Comm	307	804	23
A+B Sat (Base)-Comm	8456	1770	57
B Sat MIT	618	1336	39
MEO Sat Transfer	379	1007	29

13.3 Recommendations/Issues

The vehicle's orbit average power provides the best indication of the thermal control heater power requirements and thermal control total mass. At low spacecraft power dissipations, thermal control subsystems are extremely simple, relying on existing structure for radiator area and using bulk spacecraft temperatures to keep equipment within a nominal temperature range. As the spacecraft power increases, dissipated power densities will increase, leading to added complexity in the thermal control subsystem. Thermal doublers and heat pipes may be required to spread localized power dissipation, dedicated radiators may be needed to reject high heat loads to space, and additional heater power may be necessary to keep equipment within allowable temperatures. Considerable mass may be added to the spacecraft design. The mass of these systems are roughly 8 to 9% of the spacecraft dry mass as compared to the standard thermal control system of 3 to 5% of the spacecraft dry mass. The thermal subsystem can be refined once a design layout is selected and iterated between payload power, design integration, structures, and thermal.

14. Acronyms

ABI	Advanced Baseline Imager
ACE	Attitude Control Electronics
AD	Attitude Determination
ADACS	Attitude Determination and Control Subsystem
BBAU	Baseband Assembly Unit
C&DH	Command and Data Handling
CCD	Charged Couple Device
CDC	Concept Design Center
CM	Center-of-Mass
CP	Center-of-Pressure
DCS	Data Collection System
DOD	Depth-of-Discharge
E/W	East/West
EHS	Emissive Hyperspectral Sounder
EOL	End-of-Life
EPS	Electrical Power Subsystem
FDS	Full Disk Sounder
FEC	Forward Error Correction
FOV	Field-of-View
GEO	Geosynchronous-Earth Orbit
GMS	Geostationary Microwave Sounder
GOES-R	Geostationary Operational Environmental Satellite-R
GRB	Global Rebroadcast
HES	Hyperspectral Environmental Suite
HRG	Hemispherical Resonating Gyros
MEO	Medium-Earth Orbit
MFS	Multi-Function Sensor
MLI	Multi-Layer Insulation
MMH	Monomethylhydrazine
N/S	North/South
NOAA	National Oceanic and Atmospheric Administration
NTO	Nitrogen Tetroxide
NWS	National Weather Service

PAF	Payload Attach Fitting
RF	Radio Frequency
RHPPC	Rad-Hard Power PC
RHS	Reflective Hyperspectral Sensor
RW	Reaction Wheels
SAR	Search And Rescue
SCU	Signal Conditioning Unit
SEM	Space Environment Monitor
SIRU	Space Inertial Reference Units
SSPA	Solid-State Power Amplifier
SST	Space Segment Team
SXI	Solar X-ray Imager
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking, and Command
TWTA	Traveling Wave Tube Amplifier
W	Watts